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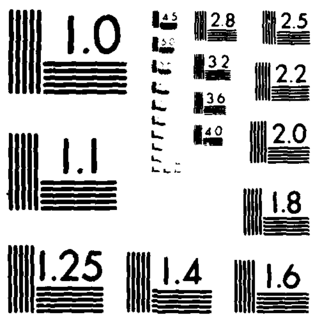

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SXTF DESIGN SUPPORT STUDY

General Electric Company
Space Division
P.O. Box 8555
Philadelphia, Pennsylvania 19101

27 February 1981

Final Report for Period 1 April 1980—27 February 1981

CONTRACT No. DNA 001-79-C-0410

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REPORT DOCUMENTATION PAGE

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1. REPORT NUMBER: DNA 5672F

2. GOVT ACCESSION NO.: AD-A110 034

3. RECIPIENT'S CATALOG NUMBER:

4. TITLE (and Subtitle): SXTF DESIGN SUPPORT STUDY

5. TYPE OF REPORT & PERIOD COVERED: Final Report for Period 1 Apr 80—27 Feb 81

6. PERFORMING ORG. REPORT NUMBER:

7. AUTHOR(s):
C. Allen, G. Haverly, R. Shoulberg, O. Allen, R. Nason, R. Schmidt, D. Gordon, H. O'Donnell, J. Peden

8. CONTRACT OR GRANT NUMBER(s): DNA 001-79-C-0410

9. PERFORMING ORGANIZATION NAME AND ADDRESS:
General Electric Company, Space Division
P.O. Box 8555
Philadelphia, Pennsylvania 19101

10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS:
Subtasks G37LAXYX959-09 and G37LAXYX960-35

11. CONTROLLING OFFICE NAME AND ADDRESS:
Director
Defense Nuclear Agency
Washington, D.C. 20305

12. REPORT DATE: 27 February 1981

13. NUMBER OF PAGES: 144

14. MONITORING AGENCY NAME & ADDRESS (if different from Controlling Office):

15. SECURITY CLASS (of this report): UNCLASSIFIED

15a. DECLASSIFICATION DOWNGRADING SCHEDULE: N/A

16. DISTRIBUTION STATEMENT (of this Report)

Approved for public release; distribution unlimited.

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

This work sponsored by the Defense Nuclear Agency under RDT&E RMSS Codes B32379462 G37LAXYX95909 H2590D and B32380462 G37LAXYX96035 H2590D.

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

SXTF	Suspension	Hardness Verification
Survivability	System Test	Vacuum Chamber
Thermal Analysis	RF Coupler	Satellite Powering

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The purpose of this study was to develop concepts and requirements to guide the development of DNA's Satellite X-ray Test Facility, SXTF. Emphasis was placed on evaluation of special concerns in those areas of the SXTF design which would directly influence a satellite manufacturer and potential user of such a facility. The evaluation and concept development was performed in seven areas:

20. ABSTRACT (Continued)

1. Satellite thermal environment,
2. Satellite support system,
3. Satellite survivability verification requirements
4. Satellite system testing
5. Vacuum test chamber design
6. Satellite powering
7. Satellite communications.

In each of these areas, recommendations are made based on previous satellite test experience which are relevent to the development and final design of the SXTF into a versatile combined effects system test facility for hardness verification.

SUMMARY

The purpose of this study was to develop concepts and requirements to guide the development of DNA's Satellite X-ray Test Facility, SXTF. Emphasis was placed on evaluation of special concerns in those areas of the SXTF design which would directly influence a satellite manufacturer and potential user of such a facility. The evaluation and concept development was thus performed in seven areas: 1) satellite thermal environment, 2) satellite support system, 3) satellite survivability verification requirements, 4) satellite system testing, 5) vacuum test chamber design, 6) satellite powering, and 7) satellite communications. In each of these areas, recommendations are made based on previous satellite test experiences which are relevent to the development and final design of the SXTF into a versatile combined effects system test facility for hardness verification.

The thermal environment of the SXTF was reviewed with the conclusion that the thermal presence of the MBS and the survivability instrumentation requirements will result in a limitation of the simulation environment and conflict with the normal thermal balance conditions required for thermal balance acceptance type testing of flight satellites. Evaluation of the thermal coupling of the MBS shield with an operating satellite test object and the room temperature MBS resulted in a requirement for a high emissivity surface facing the inside of the chamber to maximize coupling with satellite and cold walls, and a low emissivity surface to minimize any coupling with the room temperature chamber wall. While a full LN_2 environment is necessary and sufficient for operational testing of geosynchronous type satellites, a zonal control of the cold walls with up to 10 zones may be preferable for operational testing of low-orbit type satellites.

Six concepts were evaluated as candidates for supporting or suspending a test satellite in an SXTF environment. Environment, electrical isolation, reliability, remote control flexibility and operating costs were considered in the identification of a preferred suspension and support concept.

Satellite survivability verification requirements were evaluated in order to establish some baseline definition of the test orientations, instrumentation requirements and their relevance to SXTF test levels. A vulnerability assessment measurement matrix was defined for major satellite subsystems.

A comprehensive satellite systems test program was outlined for various types of satellites from research and development to production vehicles which might be tested at SXTF. Satellite and data handling approaches were evaluated along with recommendations of EAGE support. Chamber pumping systems and materials/contamination concerns were also reviewed from a satellite point of view.

Satellite powering and battery-recharging is of major concern to a satellite test program under thermal vacuum conditions. The test requirements imposed on a potential system were compared against several candidate concepts with a recommendation of a prime system.

Communication to the satellite using an RF link going through the satellites TT&C and communications payload subsystem was evaluated using several candidate RF antenna coupler designs. The designs were evaluated for their application to SXTF in the areas of frequency band, interference with the test environment and coupling characteristics leading to a hybrid concept for broader application to the SXTF.

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ACRONYM LIST

ACE	Attitude Control Electronics
AEDC	Arnold Engineering Development Center
AGE	Aerospace Ground Equipment
BFN	Beam Forming Network
CTU	Command Telemetry Unit
DNA	Defense Nuclear Agency
DSCS III	Defense Satellite Communication System, Phase III
EAGE	Electronic Aerospace Ground Equipment
ECEMP	Electron Caused Electromagnetic Pulse
JLE	Jammer Location Electronics
LNA	Low Noise Amplifier
LN_2	Liquide Nitrogen
MBA	Multibeam Antenna
MBS	Modular Bremsstrahlung Source
POE	Point of Entry
PRS	Plasma Radiation Source
RF	Radio Frequency
RLM	Receive Level Monitor
RTU	Remote Telemetry Unit
S/A	Solar Array
SCT	Single Channel Transponder
SGEMP	Systems Generated Electromagnetic Pulse
SHF	Super High Frequency
S/S	Subsystem
STARSAT	SGEMP Test and Research Satellite
SXTF	Satellite X-Ray Test Facility
TDAL	Tunnel Diode Amplifier Limiter
TLS	Transmit Level Sensor
TREES	Transient Radiation Effects on Electronics
TT&C	Telemetry, Tracking and Command
TWTA	Traveling Wave Tube Amplifier
VSWR	Voltage Standing Wave Ratio
WOD	Word of the Day

SECTION 1

INTRODUCTION

1-1 PROGRAM OBJECTIVE.

The purpose of this study program was to support DNA in the design and development of requirements for the Satellite X-ray Test Facility, SXTF. The emphasis has been placed on the translation of the operation and test facility requirements of the DSCS III Communication Satellite, the STARSAT Test and other major satellite programs into design criteria for the evolution of the SXTF.

The work described in this report was performed in seven individual task areas which directly relate to a satellite manufacturer and potential user's requirements on the design of such a test facility. These areas are defined below.

a) Thermal Environment

b) Support System

c) Survivability Requirements

d) System Testing

e) Chamber Design

f) Satellite Power

g) Satellite Isolation/Communication

Two task areas supported under this program but not described here involved support of DNA in the comparison and site selection of an appropriate vacuum facility for the SXTF and low level current injection testing on STARSAT to evaluate the correlation between free space and vacuum chamber testing.

This report is divided into seven sections, one for each major task area. In each of these areas recommendations are made, based on previous satellite test experience, which are relevant to the development and final design of the SXTF into a versatile combined effects system test facility for hardness assessment.

SECTION 2

THERMAL ANALYSIS

2-1 COMBINED THERMAL TEST CAPABILITY.

The initial consideration in this study was given to evaluating the facility issues and considerations which would be required in order to adequately perform spacecraft thermal qualification testing in the MARK 1 Facility (e.g. thermal vacuum, thermal balance) in addition to the photon radiation test thermal requirements. The performance of a thermal balance (i.e., simulation of a specific spacecraft orbit thermal environment) for a synchronous orbit spacecraft requires the addition of a solar simulation to the MARK 1 facility, with high spectrum, uniformity and collimation capability. A spacecraft thermal balance qualification involves testing of a satellite with prime coatings, precise radiator exposure, thermally isolated support, and prime properly placed thermal blankets. The instrumentation characteristics and physical presence of multiple SGEMP and TREES sensors, their leads and resulting extra penetrations needed for survivability verification in an x-ray test would be incompatible with the spacecraft configuration required for a qualification thermal balance test.

This incompatability in the requirements for the vehicle configuration and instrumentation would necessitate a chamber repressurization and a major reconfiguring and rework of instrumentation harnessing, probable insulation blanket changes and possible support system configuration changes between the x-ray test and the thermal balance test. In addition, the thermal presence of the MBS and PRS would impact on any thermal balance testing of a test object in the form of 1) increased inaccuracies

in the spacecraft thermal balance environment simulation, and 2) constraints on the allowable orientation placement of the spacecraft relative to the MBS and PRS. Thus, the facility requirements for accomplishing a realistic qualification thermal balance test are much more stringent than the requirements to provide a thermally tolerable environment for the spacecraft during SXTF testing. Table 2-1 contains a summary of the primary facility requirements and considerations for thermal control of a qualification vehicle for three different thermal test modes.

The first test objective (Column 1) addresses the performance of x-ray exposure to a selected surface of the test vehicle within a limited time period constrained to the systems transient warm-up period. This test method may be adopted if the required levels of x-ray fluence dictate close in MBS-to-vehicle surface with full-up system operation.

The second test object (Column 2) corresponds to the performance of x-ray testing on selected surfaces of the test vehicle during continuous (stabilized) operation. The operational mode(s) of the vehicle may be thermally limited to less than full-up system operation depending primarily upon the vehicle-to-MBS thermal coupling.

The third test objective (Column 3) summarizes the primary test requirements and considerations involved in providing a test facility and procedures for performing both x-ray and vehicle thermal balance testing.

2-2 THERMAL ASSESSMENT OF THE MBS.

The thermal control facility requirements and capabilities for performing just x-ray testing on an operating qualification spacecraft within the AEDC, MARK I Thermal Vacuum ($\leq 10^{-5}$ torr) Facility are much less severe than those required for a combined x-ray and thermal balance test capability.

The DSCS III spacecraft, for example, has been thermally designed for operation in a space vacuum thermal environment ranging from a negligible solar and earth radiation heating of any of its surfaces to conditions of direct solar heating, at insolation angles up to 23.45° of either its north or south radiator panels. Continuous "tolerable" temperature conditions can be maintained in the DSCS III spacecraft for x-ray testing in the MARK I Facility, if the chamber nitrogen cryogenic panels are maintained with liquid nitrogen (LN_2). Solar simulation is not required for maintaining these tolerable spacecraft thermal conditions during x-ray testing. The spacecraft thermal design incorporates multiple component and subsystem "control" heaters which will permit many modes of operation, including a partial transponder complement operation and a variety of communication modes while in this LN_2 cryo cooled chamber environment.

Elements of the Multiple Bremsstrahlung Source(e.g. debris shield and collar), are sufficient in size and are operationally temperature constrained such that they may significantly interact with spacecraft thermal radiation to the LN_2 design load. This "thermal presence" of the MBS elements may thermally limit:

1. The closure distance between the MBS elements and the major system radiators such as DSCS III's south or north radiator panels.

Table 2-1. SXTF thermal control capability considerations.

($\leq 10^{-5}$ TORR)

	I	II	III
TEST OBJECTIVES	X-RAY TESTING WITH TRANSIENT THERMAL LIMITS (e.g. STARSAT)	X-RAY TESTING UNDER STEADY STATE OPERATION	COMBINED X-RAY AND THERMAL BALANCE TEST CAPABILITY
TEST FACILITY	THERMAL VACUUM LN_2 (1 ZONE)	THERMAL VACUUM LN_2 (1 ZONE) POSSIBLE LOCAL HEATING FOR OPERATIONAL MODES $\leq$ FULL-UP	THERMAL VACUUM LN_2 (1 ZONE) SOLAR SIMULATION SPECTRUM, UNIFORMITY, COLLIMATION POSSIBLE LOCAL HEATING FOR OPERATIONAL MODES FULL UP
CONSIDERATIONS	TEST TIME WINDOW LIMITED	TEST VEHICLE MAY HAVE TO BE OPERATED AT LESS THAN FULL-UP MBS & PRS THERMAL PRESENCE LIKELY COMPENSABLE EXCEPT FOR "CLOSE-IN" MBS TESTING	MAJOR TEST VEHICLE RECONFIGURATION FOR THERMAL BALANCE TESTING MBS & PRS THERMAL PRESENCE CONSTRAINING TEST ACCURACIES AND/OR ORIENTATION FOR THERMAL BALANCE TESTING

2. The system operation modes to reduce conditions of dissipation in satellite payload or TT&C subsystems.

3. The period of operation of the satellite's major subsystem power radiator assemblies from full up system operation.

The current SXTF operations and procedures definition indicates that "all spacecraft subsystems should be operational during the tests" and "that close-in MBS exposures .... be performed" on selected spacecraft surfaces. The preliminary design status of the MBS debris shield, collar and module operational temperatures are in their initial design definition state. The debris shield in the current definition appears to be the primary MBS element in the thermal environment of an adjacent vehicle surface. A proposed "strawman design" for the shield consists of multiple layers of Kapton or Mylar with a thin layer of aluminum. The proposed design places the aluminum layer facing the test vehicle with an identified thermal optical emissivity of 0.2.[1]

A thermal evaluation has been made of the effect of placing the DSCS III North Panel at a distance of about 9 feet from this "strawman" debris shield with an outer layer design of $\epsilon = 0.2$. This distance corresponds to the approximate location of the DSCS III spin axis at the center of the MARK I facility and an MBS shield and collar within about 1 meter proximity to the facility pressure wall. This simplified evaluation indicates that for a full-up system operation (i.e., 502 watts thermal dissipation) of the DSCS III North Panel an over-temperature condition would result for any initial temperature of the debris shield as shown in Figure 2-1.

A similar evaluation was made assuming a modification to the shield such that a more efficient thermal decoupling was achieved by the use of the characteristic emissivity ratio (i.e., vehicle facing/MBS facing surfaces emissivities) of 0.65/0.2. An evaluation was made of this MBS shield configuration and the test chamber conditions illustrated in

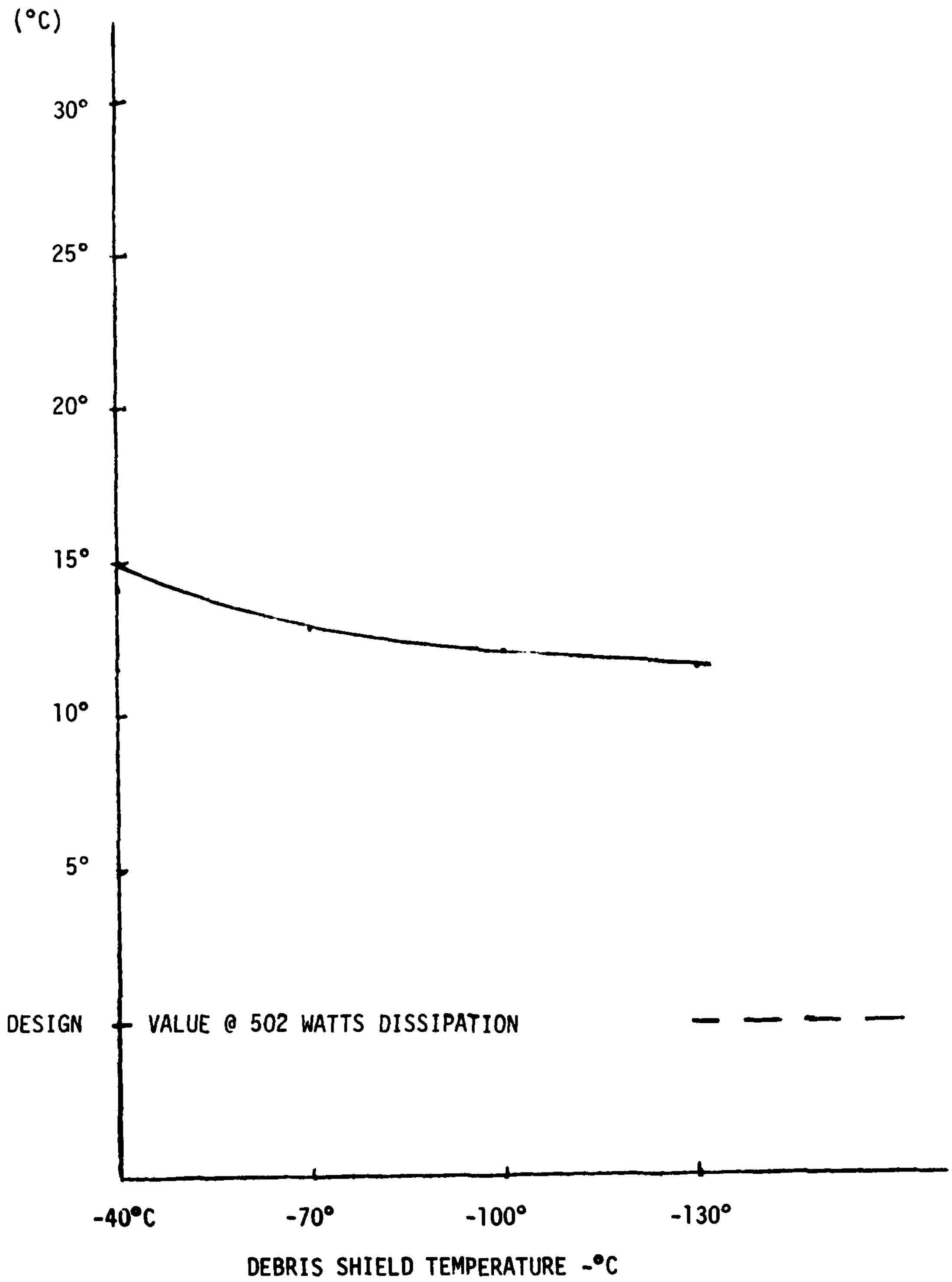


Figure 2-1. DSCS III north panel average temperature.
Excess of Design

Figure 2-2 for two spacecraft positions. The predicted average north panel steady state temperatures for test conditions are approximately 10°C under and 10°C over the design temperature, respectively, for the two spacecraft positions in the center of the chamber (#1) and close in to the MBS (#2). These predictions of average north panel temperatures are approximate for the purposes of guiding SXTF preliminary test facility design issues.

The DSCS III north panel is a complex assembly of electronic components with differing temperature constraints, and significantly differing thermal dissipation densities and it is integrally coupled with the entire spacecraft thermal balance. The determination of the specific satellite environmental limitations on placement of the DSCS III spacecraft surfaces relative to the MBS elements must be determined by a detailed spacecraft thermal evaluation. This evaluation must evaluate the entire spacecraft subsystem thermal balance considering the influence of the shield presence on all spacecraft surfaces and the individual component characteristics and temperature constraints. The average panel temperature should not be used for anything other than a qualitative evaluation since it may be off by up to 20°C above or below individual critical element temperatures.

It appears likely that test requirements will be encountered which will advocate placement of the DSCS III North or South Panels relatively close to the MBS to achieve a higher fluence x-ray energy deposition. If full-up spacecraft system operation is a necessary test condition with such major MBS shield radiation blockage of either of its (North or South) Radiator panels, the operation period will be limited to prevent over temperature,

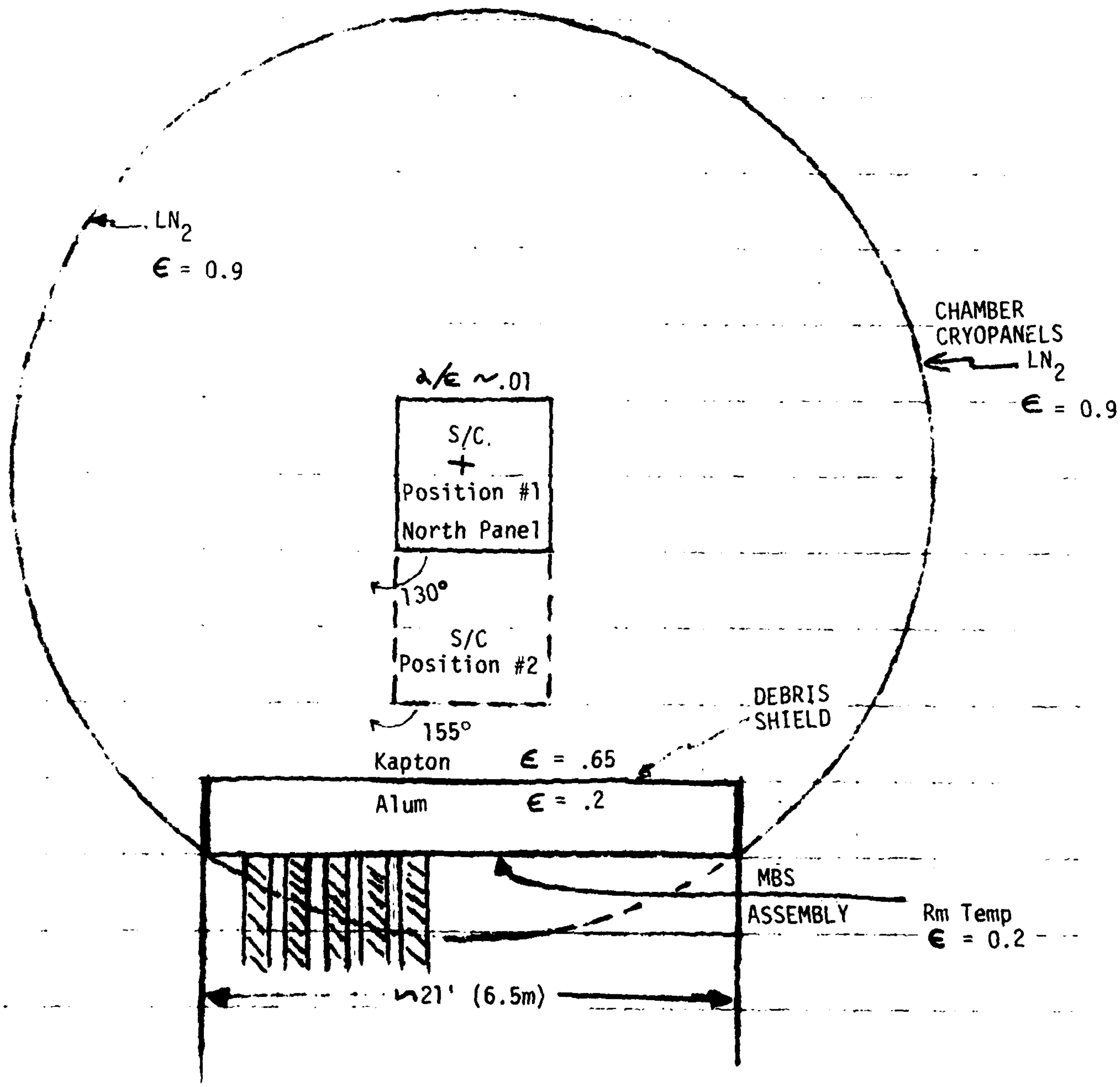


Figure 2-2. Spacecraft thermal environment.

and probable degradation of its electronic components. The x-ray test objectives may be met for such conflicting spacecraft operational and test facility environment conditions by performing the x-ray testing in conjunction with a transient (e.g., time limited) spacecraft operation mode. The DSCS III spacecraft could be maintained in its minimum dissipation (e.g., survivial) mode and "pre-soaked", to a stablilized thermal condition, in the MARK I LN_2 cooled facility with all radiation elements maintained at minimum temperature conditions. X-ray testing could possibly follow switching of the system to its full-up operational mode. However, from discussions in Section 5 the minimum switching or turn-on time required may have a serious impact on the thermal temporal profile of the satellite and, therefore, on the allowable test mode.

2-3 SATELLITE TYPE.

The DSCS III spacecraft is a geosynchronous three-axis stabilized communications satellite incorporating an integrated system thermal control design. The test facility requirements for simulating a geosynchronous orbital thermal environment (or providing a tolerable thermal environment) are significantly different than those required for satellite systems having other attitude orientations, for example, near-earth altitudes or different sensor payloads and functions. As an example, the earth viewing sensors and thermal radiator surfaces for near-earth orbiting or surveillance satellites, such as LANDSAT-D, are dependent upon specific design levels of earth reflected and radiated energy. The test facility must include a means for local heating so as to maintain a tolerable thermal vacuum environment for these radiator and sensor surfaces. Modification of the MARK I facility to permit multi-zonal control of the cryo panels with controlled GN_2 capability appears to be one means for providing differing levels in a spacecraft local thermal environment. Separate zonal control of the cryogenic system consisting of 10 zones: four quadrants of the cylindrical panel above and below the chamber equator plus the ceiling and floor should provide a sufficiently tolerable thermal environment for most low orbit satellites.

SECTION 3

SUPPORT SYSTEM

3-1 *INTRODUCTION.*

This support system trade study for the proposed SXTF at the AEDC Mark I chamber focused on two methods of supporting a test vehicle in the facility, namely:

1) A gimballed rigid support fixture, and
2) A cable/rope suspension system similar to that used for STARSAT in the Huron King experiment.

Either would have the capability to reposition the vehicle.

The study also examined methods of supporting vehicle appendages, such as deployed solar arrays, as trade-off factors. The DSCS III, a geosynchroneous orbit vehicle, was used as a baseline for comparison of suspension system issues. However, consideration was given to the capability of the support system to handle future larger classes of vehicles, like GRO, and low earth orbit type vehicles like Landsat D, and NOSS.

3-2 SUPPORT SYSTEM REQUIREMENTS.

This section presents the requirements of the system. Resulting from these requirements, there are also implications and limitations which are added comparison factors affecting the recommendation of this study.

Due to operation or use of any SXTF support system in the combined vacuum, thermal, electrical and radiation environments anticipated for SXTF; there are a number of unique requirements or issues which need to be addressed in any comparison of candidate systems. These include:

1. Vehicle Types
2. Configurations

3. Repressurization Frequency
4. Positioning Accuracy
5. Power Isolation
6. Electrical Isolation
7. Thermal Isolation
8. Materials and Contamination

The following paragraphs discuss the major concerns in these areas.

Vehicle Types - The DSCS III vehicle, being used as a baseline in this study, is a member of a class of 3-axis stabilized defense communication satellites designed for geosynchronous orbit operation. It is shown in its orbital configuration in Figure 3-1. With vehicles already in the development stage in the 30,000 pound class, such as NOSS and GRO, it is reasonable to consider from a size/weight viewpoint, future vehicles which may use the SXTF. Future synchronous defense application vehicles may be in the size/weight class of GRO (Gamma Ray Observatory). Low earth orbit type vehicles might be in the Landsat D, DMSP, or NOSS class. An orbital configuration drawing is shown in Figure 3-2 for Landsat D, while conceptual drawings of GRO and NOSS are in Figures 3-3 and 3-4. The requirement for consideration of these larger satellites is driven by the advent of the increased launch capability of the Space Shuttle as shown in Table 3-1. These dimensions represent a reasonable specification requirement on the size of a centerbody support system.

Spinner type satellites that would use the SXTF also have good potential for growth in weight and size. However, they tend to be less restrictive in their width requirements and would be confined by only the size of the shuttle bay.

Configurations - Figure 3-5 shows the significant features of the AEDC Mark I Chamber which is proposed to be modified for the SXTF. The 36 foot inside diameter is the feature which most influences the possible configurations of a test vehicle and its appendages. In addition,

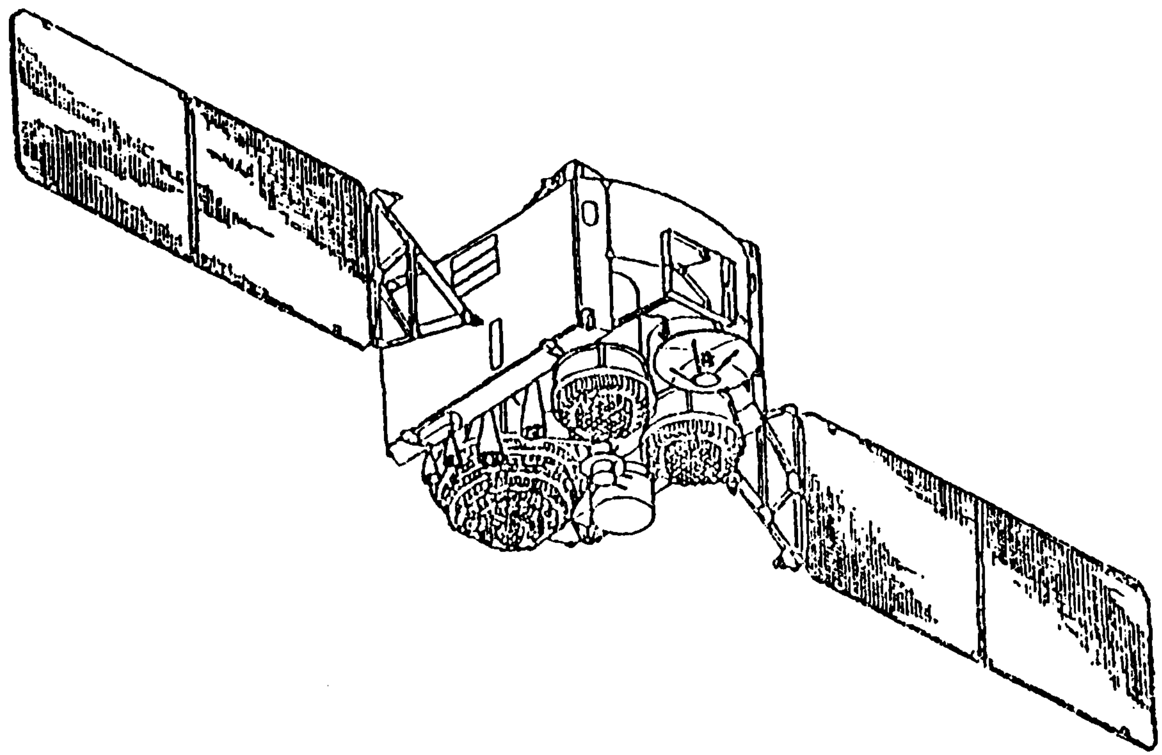


Figure 3-1. DSCS III orbital configuration.

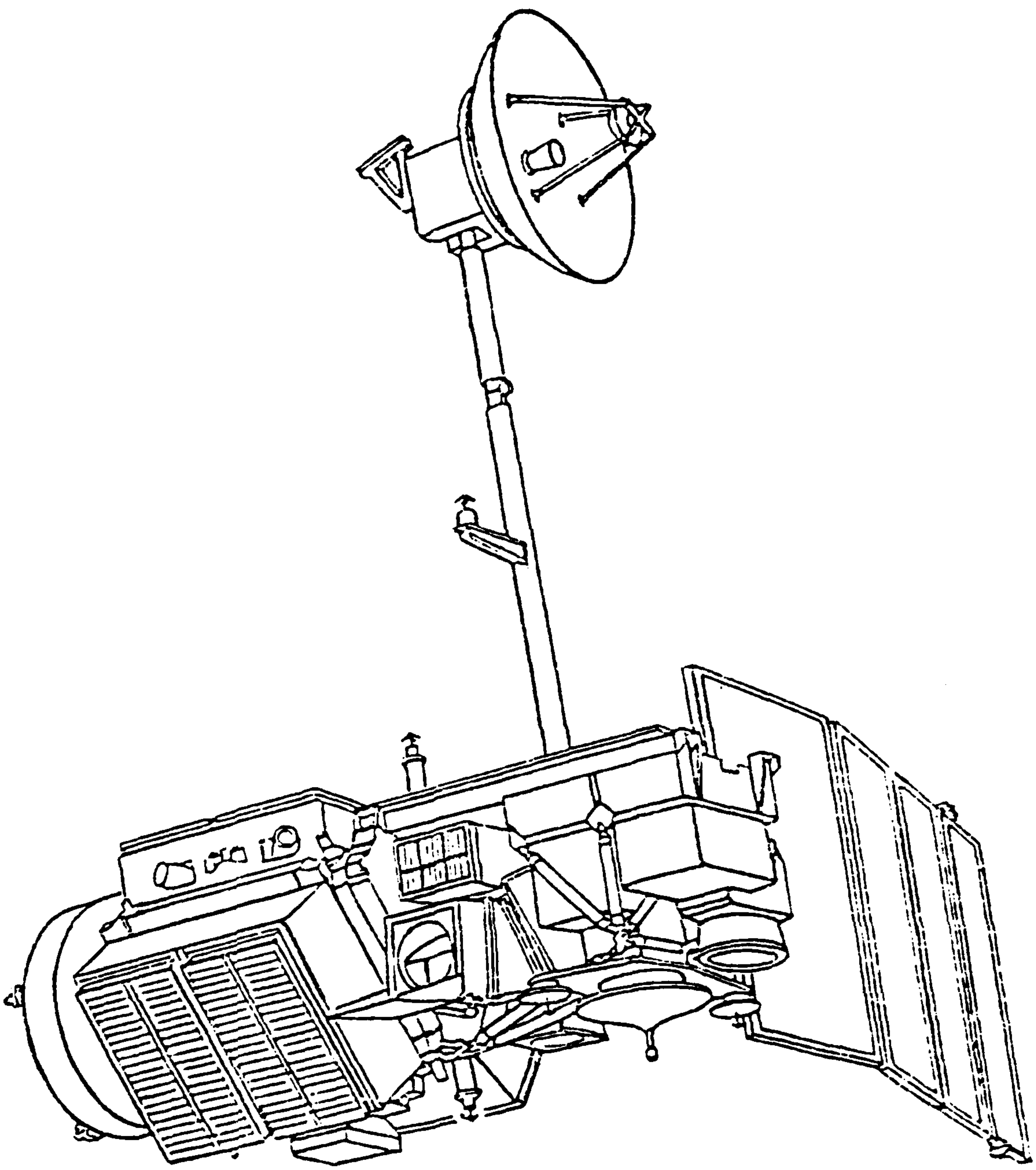


Figure 3-2. Landsat D orbital configuration.

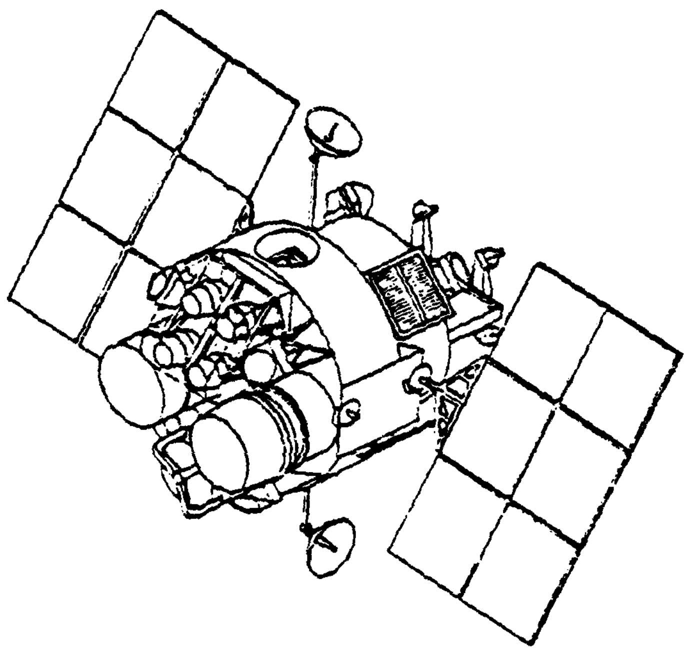


Figure 3-3. GRO conceptual orbital configuration.

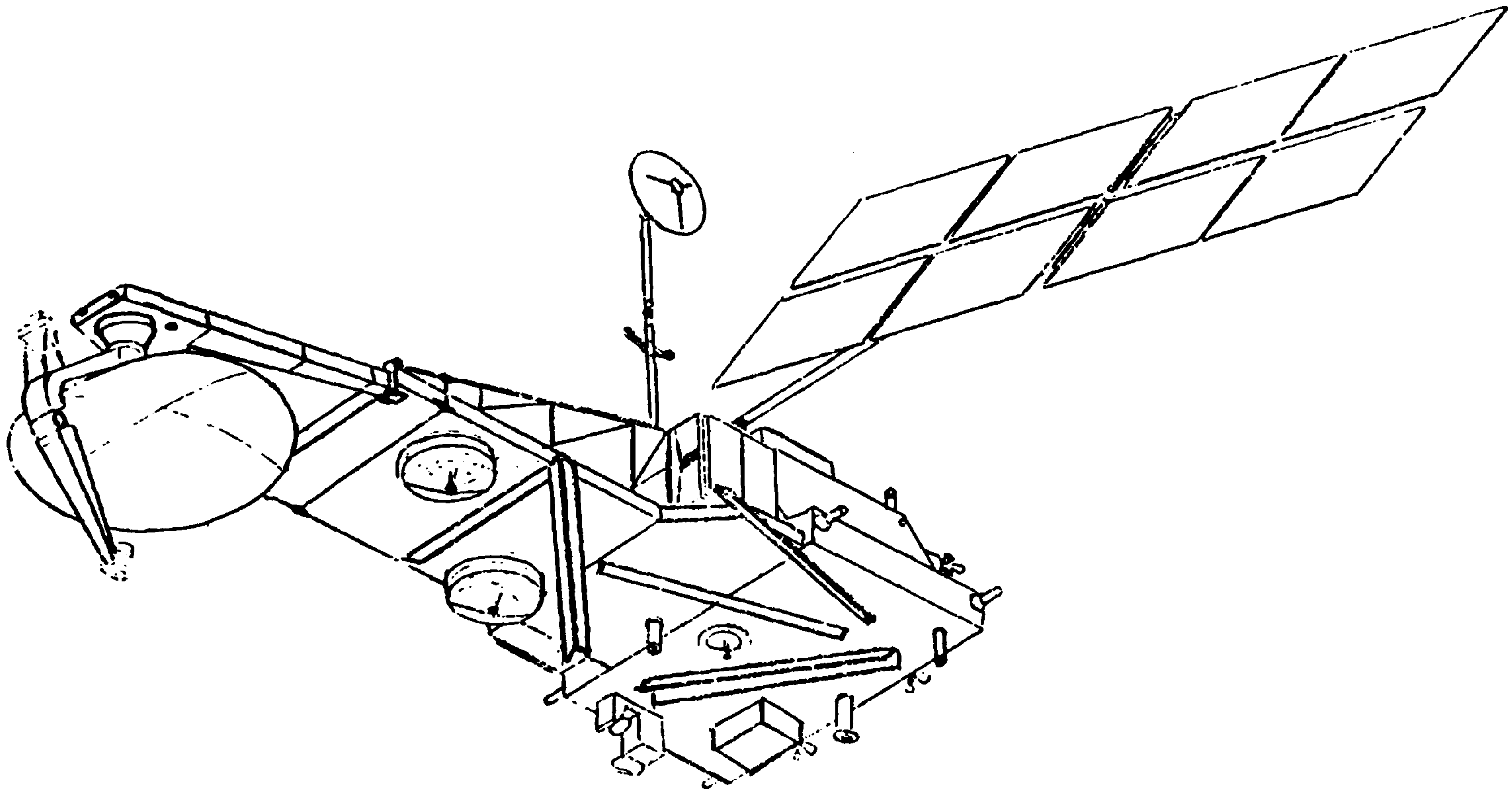


Figure 3-4. NOSS conceptual orbital configuration.

Table 3-1. Future satellite characterization.

- SHUTTLE

  PAYLOAD BAY DIMENSIONS

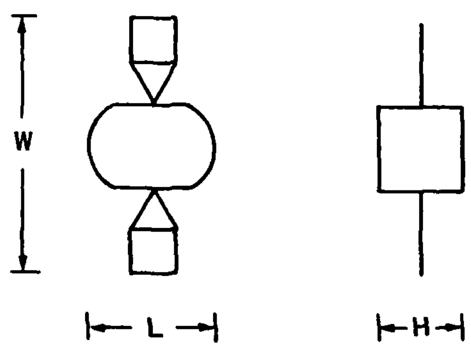
  LENGTH - 60 FT.

  DIAMETER - 15 FT.

  MAXIMUM LAUNCH PAYLOAD WEIGHT

  65,000 LBS.

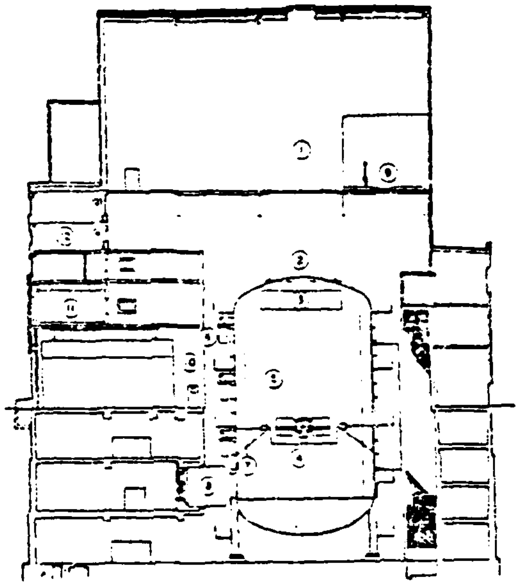
- CURRENT AND FUTURE SPACECRAFT



DIMENSIONS WITH APPENDAGES DEPLOYED

	L-FT	W-FT	H-FT	WT-LB
LANDSAT - D	14	27	19	3900
DSCS III	11	38	7	2357
NOSS	45	42	33	14000
GRO	23	43	20	32000

1 BUILDUP AREA
2 MAIN CHAMBER
3 SOLAR SIMULATOR
4 TEST ARTICLE
5 TEST ARTICLE HANDLING SYSTEM
6 DIFFUSION PUMPS
7 COLD WALL & CRYOPUMPS
8 ACCESS LOCK
9 CLEAN ROOM
10 MAIN BUILDING ENTRANCE
11 CONTROL ROOM
12 DATA ACQUISITION ROOM



- VACUUM CHAMBER SIZE: 42' DIAM x 82' HIGH (OUTSIDE)
  36' DIAM x 65' HIGH (INSIDE)
- PRESSURE ALTITUDE: SEA LEVEL TO 300 STATUTE MILES
  (1 x 10[illegible] TORR)
- THERMAL RADIATION SIMULATION: SOLAR (12' x 18');
  ALBEDO; EARTHSHINE
- WALL TEMPERATURE: 77°K (-320°F)*
- CRYOPUMP TEMPERATURES: 22°K (-423°F)**; 4°K (-452°F)***
- DYNAMIC SIMULATION: 2-SEC ZERO-G OPERATION
- PLUME TEST CAPABILITY: MAINTAIN 240,000-FT ALTITUDE
  FOR ENGINES UP TO 300-LB THRUST AND
  300,000-FT ALTITUDE FOR ENGINES UP
  TO 25-LB THRUST

*LIQUID NITROGEN, **GASEOUS HELIUM, ***LIQUID HELIUM

Figure 3-5. Characteristics of the AEDC Mark I chamber.

the modification of the chamber with an electromagnetic damper will further reduce the working diameter to 32 feet. For a DSCS III vehicle, having a fully deployed solar array with an overall length of approximately 38 feet, the support system should be capable of providing repositioning of this vehicle within the envelope of a double cone surface of revolution. The cone angle is approximately 90^{o}. This geometry is illustrated in Figure 3-6. With the 21 foot square Modular Bremsstrahlung Source (MBS) located on the vertical chamber wall, only exposure at oblique angles between the radiation source, and the North or South Instrumentation panels would be possible with the solar panels deployed. With the removal of one or both solar array paddles, however, the support system should be capable of all configurations for maximum exposure of the vehicle.

The support of the solar arrays discussed in the following section can be made using one of three approaches: 1) attachment to the suspension system, 2) attachment to the vehicle center body, or 3) independent support from chamber hard points. The choice of solar array support is driven by the requirement that the position and/or orientation of the vehicle can be remote controlled or varied between shots of the x-ray source to some degree without the need for repressurization. It is clear that as the vehicles that will use the SXTF Facility increase in size to the 10,000 pound class, a severe limitation in the number of possible configurations may develop especially if deployed antennas, booms and solar arrays are a part of the test configuration.

<u>Repressurization</u> - Closely related to the subject of configuration and the requirements for remote positioning of the test vehicle is the desire to minimize the occasions for repressurization. Because of possible vehicle test schedule constraints, the time required for vehicle check-out, and the chamber turnaround time, it would be ideal to perform the radiation tests in several of the required test orientations between chamber openings. This provides

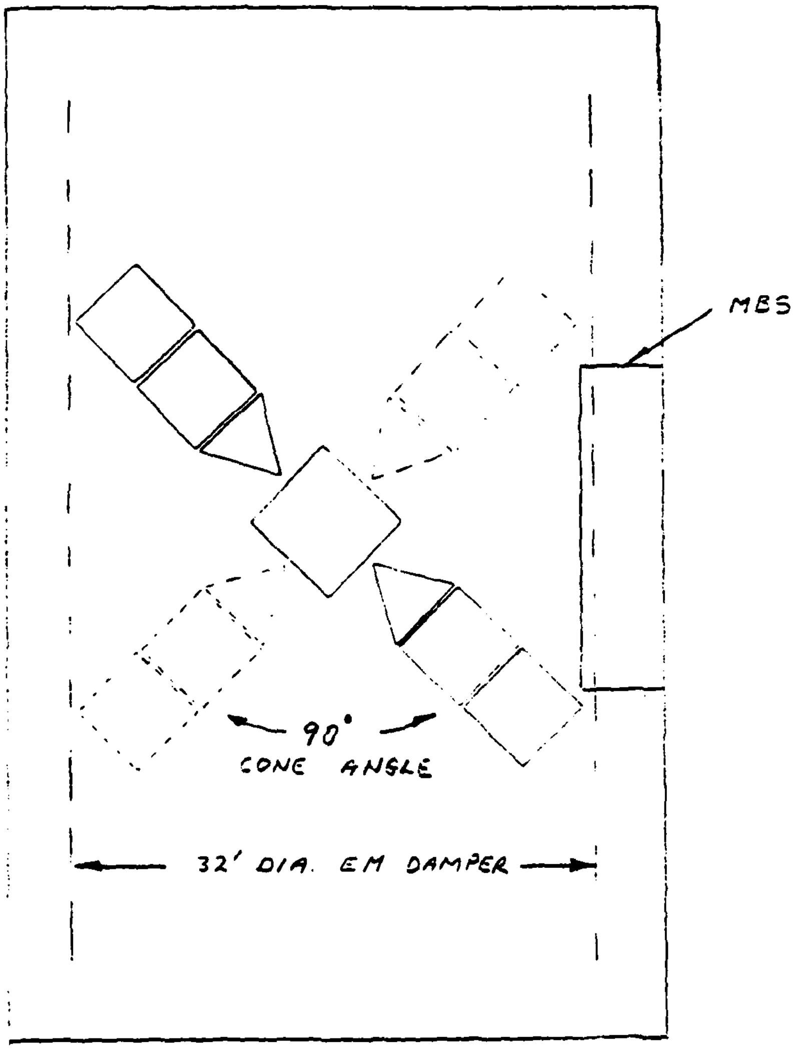


Figure 3-6. DSCS III in the AEDC Mark I chamber.

trade-off requirements on the vehicle and solar array support system which vary from total reconfiguration without chamber opening through limited reconfiguration for certain tests or sets of tests, to manual reconfiguration after each test.

Positioning Accuracy - The need for pointing capability or accuracy in the vehicle support system is dependent on two features of the SXTF: 1) the use of RF links for commanding and communicating with the vehicle, and 2) position of the vehicle with respect to the fixed radiation sources.

Three options for commanding the vehicle under vacuum are the use of RF links, fiber optics, and by a retractable umbilical EAGE. The RF link might consist of a coupler system which is fixed with respect to the vehicle and moves around with it. An alternative multi-coupler system might also be used which may require aiming the vehicle with some predetermined pointing accuracy to each coupler. The fiber optics for commanding and communicating with the vehicle, consisting of a cable from the vehicle to the chamber wall, are such that their design may not require the same degree of repositioning capability for the support system. However, the presence of such cables will have some affect on the movement and repositioning capabilities of the support system.

Power Isolation - Considerations of power isolation for the test vehicle generate requirements for the vehicle support system. If a powering umbilical is used to supply ground power and battery recharging, then the support system might only be required to provide support to the cable close to the vehicle while still maintaining some degree of repositioning capability. There are, however, other methods of supplying ground power which offer greater vehicle maneuverability and power isolation.

The support system must be capable of mounting the "receiving" unit or be adaptable to its mounting on the vehicle, and possibly permit

realignment of the "receiving" unit. In the case of mechanically retractable devices it could consist either of an easy mating electrical plug or a capacitive or inductive coupler.

Electrical Isolation - The choice of materials for the vehicle and solar array support system is the controlling factor when assessing electrical isolation requirements. The goals are to minimize high frequency SGEMP coupling to the chamber and eliminate the low frequency or DC connection to the chamber during electrostatic charging experiments. The use of dielectric materials with high electrical resistivity is therefore required for electrical isolation, and limits the structure or cabling to non-metallic materials like Bakelite, fiberglass, Kevlar, nylon, glass reinforced nylon, and fiber reinforced phenolics.

Where it becomes absolutely necessary to use metals, it is required to minimize the interface with the vehicle, and maximize the distance between metal and vehicle.

Thermal Isolation - The requirements on the solar array and vehicle support systems can be severe or reasonable depending on whether the SXTF shall have thermal balance or thermal vacuum capability.

In the cryogenic environment it is necessary to provide conductive thermal isolation between the vehicle and support system by using low thermal conductivity materials. This would apply to any possible thermal environment test mode.

Radiative isolation requires that the support systems exhibit no significant thermal presence either as energy radiators or as shadowing members for solar simulator or heat lamps or the cold walls. For low earth

orbit vehicles, this situation is made more complex by the existence of earth albedo simulation sources or zonal conditions in cryogenic systems. These restrictions apply more in the thermal balance mode, than in the thermal vacuum conditioning test mode where some shadowing may be permitted, and thermal presence is not as restrictive.

Materials and Contamination - Factors affecting the choice of materials other than those mentioned above include outgassing and degradation due to extremely low temperatures and radiation. Low outgassing is desirable. Materials which turn brittle like Teflon, or certain epoxy resins which lose structural properties with radiation are clearly undesirable.

3-3 SOLAR ARRAY SUPPORT.

If the deployed solar arrays are required during X-ray or other radiation testing in the SXTF, they will need to be supported whatever the method of vehicle support used. Generally, the options are to support them with flexible or stiff members either independently or fixed to the vehicle or vehicle support.

The requirements above form the basis for selecting the solar array or appendage support system. There are, however, features that are unique to each vehicle which must be considered. It is likely that the support suitable for the DSCS III vehicle might not be suitable for another, and an evaluation of requirements for a unique support method for the deployed appendages of each vehicle using the SXTF should be considered.

The strength allowables of the DSCS III solar array driver mechanism at the yoke interface are approximately 2800 in-lb bending, 240 in-lb torsion, 380 lb thrust, and 660 lb transverse load. The weight of the arrays is approximately 120 lbs. If design loads with a factor of safety of 2.0 are

used, then the maximum excess support system weight contributing to thrust loading is approximately 70 lbs. Therefore, any solar array support system must not allow any transfer of the vehicle load to the solar array drive mechanism either during the assembly, reorientation or test phases.

There are four approaches for supporting solar arrays shown in Figure 3-7. The configurations include both the gimbal support and suspension support approaches for the vehicle. There are other options such as an independent suspension support for the solar arrays with a gimbal vehicle support, but these are not practical for operation in a vacuum system.

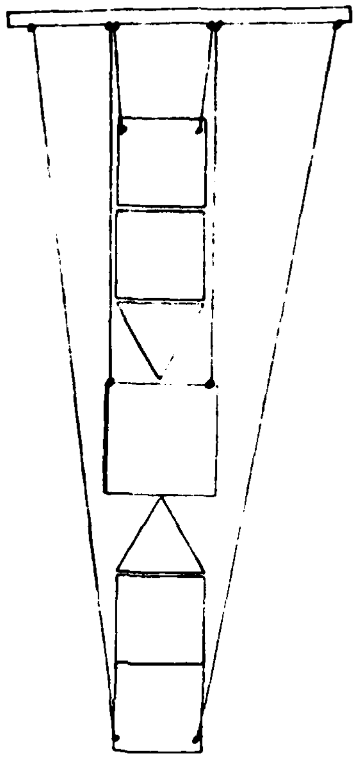
Generally, those factors favoring all the approaches and those against all approaches, are not listed. For example, the orientation or rotation of the solar arrays about the vehicle axis is not considered.

It is clear that the approaches in which the solar arrays are independently support from chamber hard points are electromechanically difficult or are restrictive from a vehicle reconfiguration standpoint. *Approach* 4, which is a lightweight combination of frame and tension support, has most advantages. All approaches suggest that support of the arrays from the vehicle center body is desirable and should be considered for reaching the design goals of the vehicle test configurations.

3-4 TEST ORIENTATIONS.

A summary of the reconfiguration requirements and the general approach characteristics for the support system is shown in Table 3-2. Figure 3-8 shows the relationship between a typical 3-axis stabilized vehicle center body and the MBS. A typical spin-stabilized test vehicle is shown in dotted lines.

From the discussions in Section 4 there are approximately six spacecraft orientations proposed for a typical test vehicle, such as the DSCS III

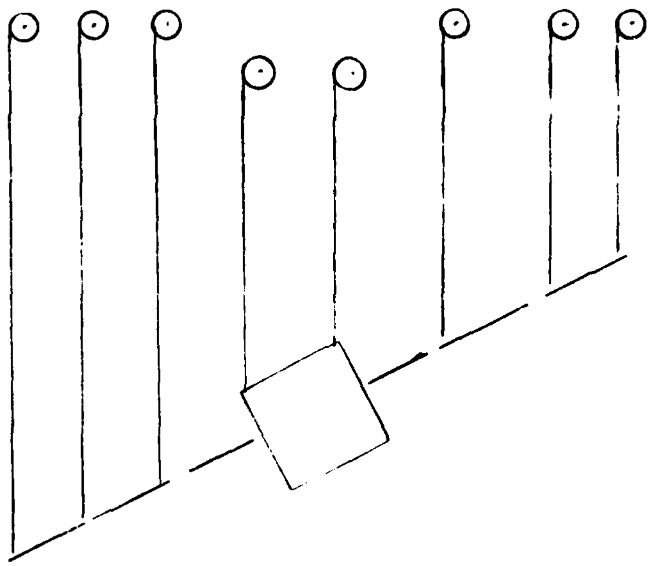


Approach 1

SOLAR ARRAY SHAFT AND HINGE SUPPORT ASSISTED BY LINES - VEHICLE SUSPENDED

PRO	CON
+ Simple arrangement	- Configurations limited to rotations about vertical.
+ Isolation good	- Handling risk. Overload hinges in bending.
	- Relative lateral motion between center body and panels.

Figure 3-7a. Solar array support candidate #1.



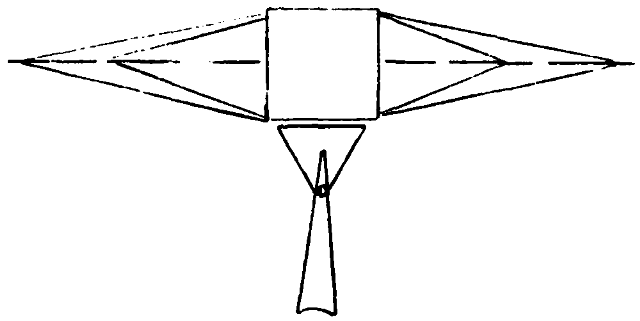
APPROACH 2

SOLAR ARRAYS INDEPENDENTLY LINE SUPPORTED - SUSPENDED VEHICLE

ALL LINES ADJUSTABLE

PRO	CON
+ Totally reconfigurable to limits of chamber.	- Difficult electro-mechanical control and coordination.
+ Good isolation.	- Handling difficult. Possible relative motion.
	- Live adjustment problem at low temperatures.

Figure 3-7b. Solar array support candidate #2.

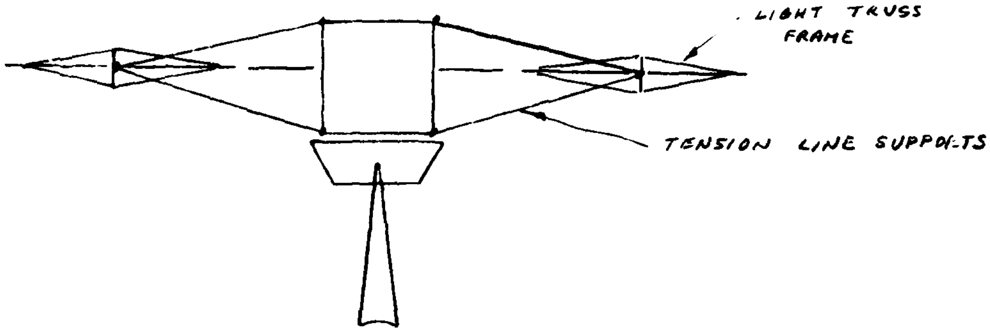


APPROACH 3

SOLAR ARRAYS TRUSS SUPPORTED, FROM THE VEHICLE - GIMBAL FIXTURE SUPPORTED VEHICLE

PRO	CON
+ No relative motion of solar arrays w.r.t. vehicle.	- Large truss members. Cumbersome.
+ Totally reconfigurable to limits of chamber.	- Potential shadowing problems.
+ Adaptable to suspension supported center body.	
+ Trusses alternately vehicle or support fixture supportable.	

Figure 3-7c. Solar array support candidate #3.



APPROACH 4

SOLAR ARRAYS FRAME AND TENSION LINE SUPPORTED, FROM THE VEHICLE - GIMBAL FIXTURE SUPPORTED VEHICLE

PRO	CON
+ Totally reconfigurable to the limits of chamber.	- Intricate structure; requires careful assembly.
+ Enhanced thermal isolation; reduced shadowing.	
+ Lightweight tension supported lines with lightweight panel supporting frame.	
+ Adaptable to suspension supported centerbody.	
+ Tension supports alternately vehicle or support fixture supportable.	

Figure 3-7d. Solar array support candidate #4.

Table 3-2. Reconfiguration capability requirements.

DEGREES OF FREEDOM	APPROACH	REMARKS	
• ROTATION ABOUT VERTICAL	CAROUSEL	EASIEST RECONFIGURATION. LEAST COMPLEX TO PROVIDE	MAJOR
• HORIZONTAL TRANSLATION ALONG MBS AXIS	RAIL GUIDED, MOTOR DRIVEN GEARS/WHEELS	PROVIDES VARYING INTENSITY RADIATION. ALLOWS REPOSITIONING FOR JOINT PRS, MBS EXPOSURE.	
• VERTICAL TRANSLATION	COORDINATED MOTORIZED HOISTS/ PLATFORM ELEVATOR	ACCOMMODATES VARIED HEIGHT VEHICLE. PERMITS SELECTIVE DEPLOYED SOLAR ARRAY/CENTERBODY EXPOSURE.	MINOR
• HORIZONTAL ROTATION	INDEPENDENTLY MOTORIZED HOISTS/ GIMBALLED PLATFORM	PERMITS COMPLETE EXPOSURE PROFILE OF SPINNERS AND CENTERBODY WITHOUT REPRESSURIZATION	

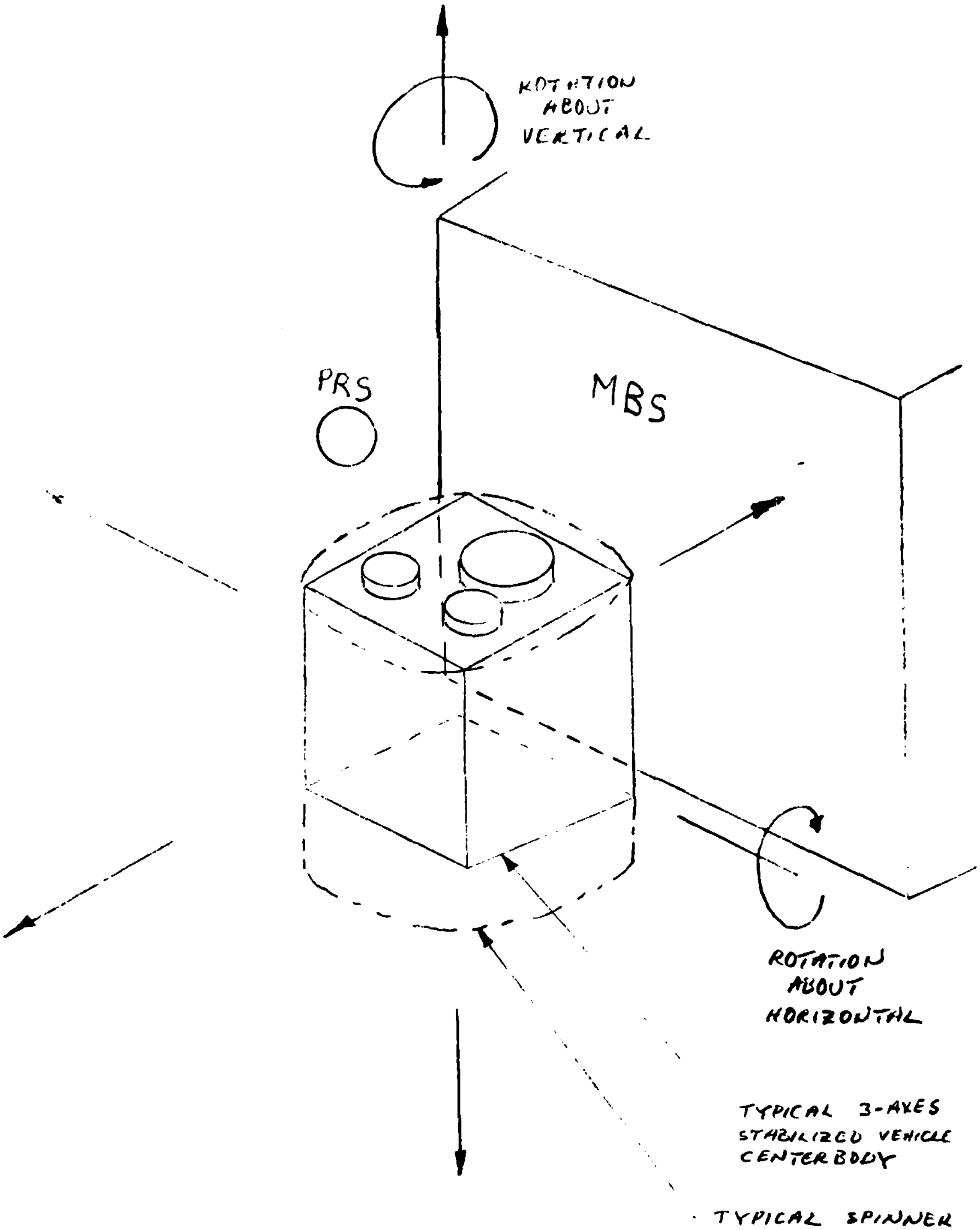


Figure 3-8. Desirable test reorientation capability.

satellite, in the SXTF. These are listed below in terms of the face and/or component directed toward the MBS:

a) East Bay

b) North Panel

c) West Bay

d) South Panel

e) Earth Face Antennas

f) Earth/Antennas with Solar Arrays Deployed or Oblique Angle with Solar Arrays Deployed.

A simple scenario which permits the above configurations would be to position the vehicle center body without the solar arrays deployed in front of the MBS Source and with antenna axes vertical. In this position orientations a, b, c, and d could be exposed with reconfiguration by a simple rotation about *the vertical chamber axis.* Rotation of the vehicle about the chamber horizontal axis would give a configuration for orientation e. Installation of the solar arrays would be possible for this orientation and also allow exposure in configuration f. In this configuration exposure of orientations a and c would also be possible with the solar arrays deployed. Testing and remote reorientation of a test vehicle at an oblique angle with its appendages, such as the solar arrays deployed would have to be considered on an individual basis because of the uniqueness of these dimensions. The above scenario suggest that rotation about the vertical is a major asset for the vehicle support system. However, in an assessment of the need for horizontal rotation capability, some specific handling and assembly issues need to be considered.

Testing the satellite with the solar arrays deployed requires that the solar array axes be essentially vertical, i.e., the North and South panels facing up and down in the chamber. This requires that the vehicle be rotated 90° from its normal environmental testing position and then attaching the

solar arrays to the vehicle.

Solar Array installation can conceivably be accomplished either in the chamber or in the preparation area. The handling requirements for the DSCS III qualification and prime vehicles are quite stringent, however, making it difficult to satisfy solar array assembly and installation conditions inside the chamber.

Since the chamber will have to be vented to change from a center body exposure of the North or South panels to a solar array's deployed configuration with assembly either in or outside the chamber, it would be preferrable to carry out vehicle horizontal rotation in the preparation area, install the solar arrays, then lower the entire assembly into the chamber. Rotation in the preparation area would be performed with a piece of MAGE, such as a turn over fixture which would rotate the vehicle antenna axis from a vertical to a horizontal position.

It is therefore suggested that horizontal rotation capability in the support system is not as important as the 'carousel' requirement. Translation along the MBS horizontal axis is, however, another major highly desirable feature of the support system because it permits varying radiation intensity with distance of the vehicle from the source. Although vertical translation capability permits accommodation of vehicles of varying height, the vertical dimension of the MBS ($\sim$21 ft) reduces the need for it.

3-5 SPACECRAFT HANDLING.

This section introduces preliminary ideas directed towards assessing problems that could arise in using the support system. A support system employing suspension lines may need to be attached to the vehicle prior to both being lowered together into the chamber. Reference to Figure 3-9 demonstrates that there are common hoisting zones for the system and for the

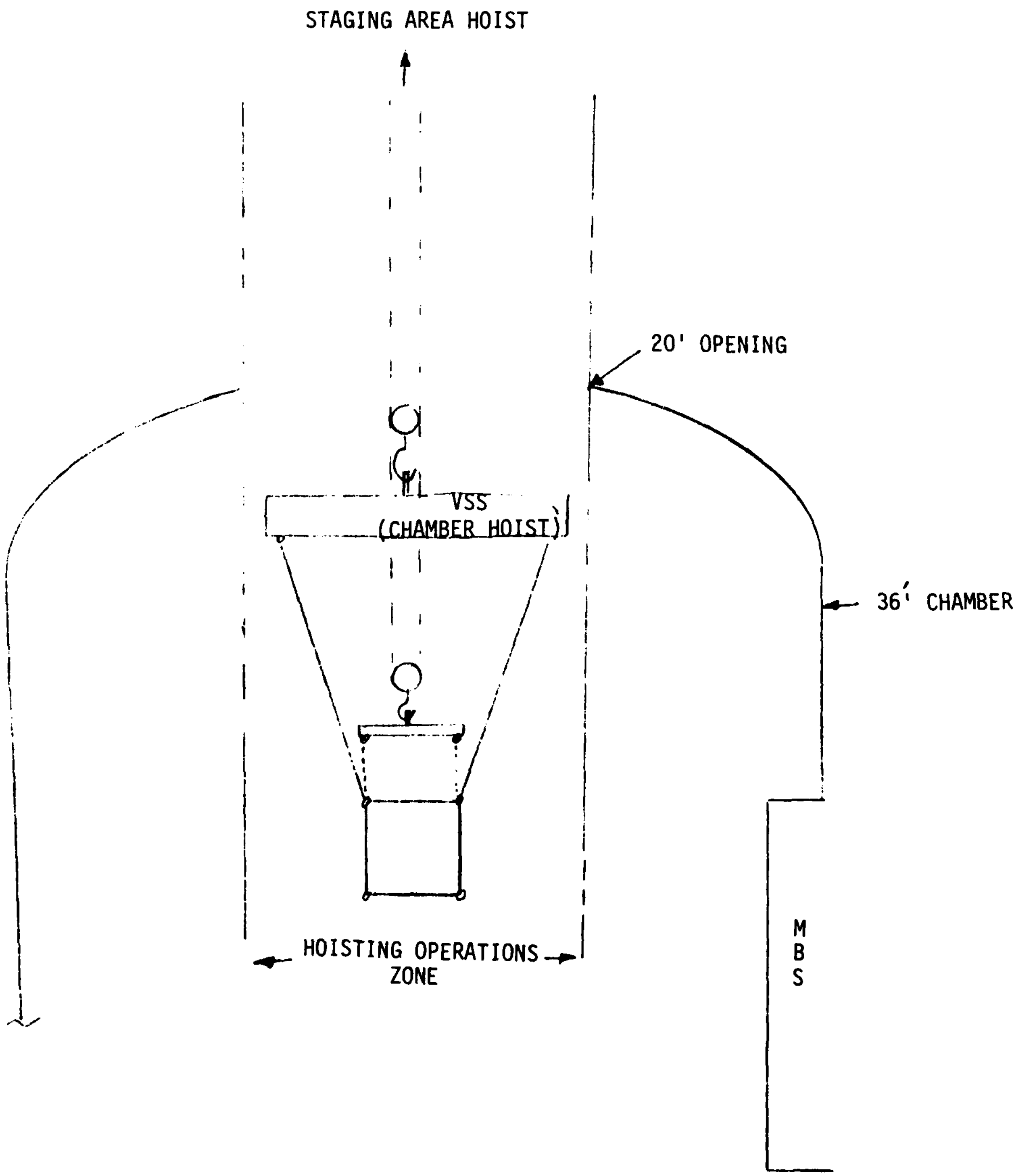


Figure 3-9. Satellite loading.

staging (prep) area hoist, since they must share the limited 20' equipment door.

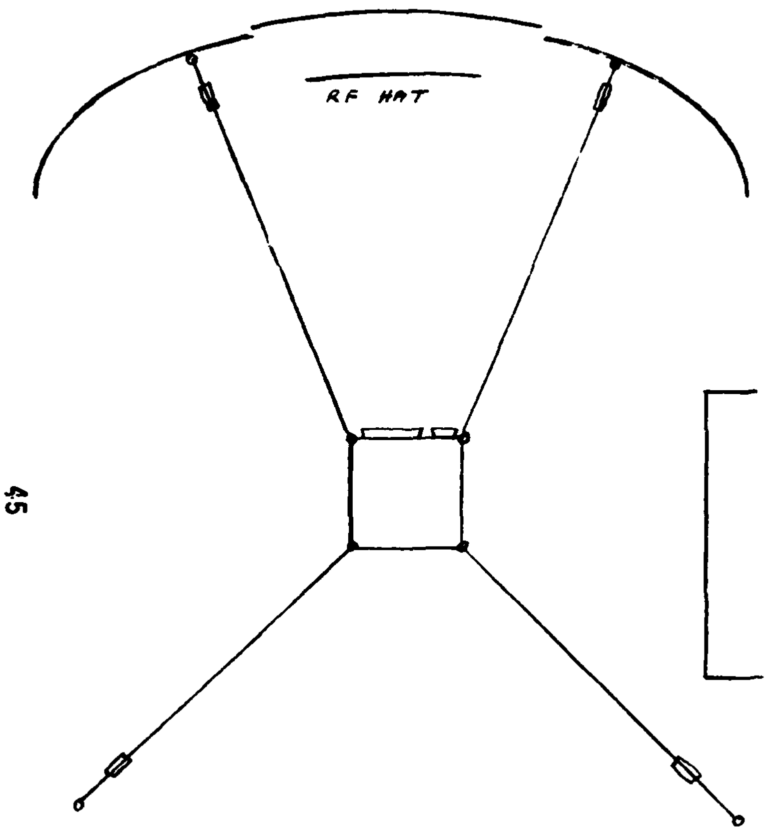
The choice of support system is thus dependent upon its compatibility with both the vehicle and solar array installation methods. Because the chamber can accommodate the deployed solar arrays in their double cone envelope (Section 3-2), their installation must either be performed inside the chamber, or in the preparation area with the spacecraft and its deployed arrays being lowered as one piece into position in the chamber. In either case, compatible MAGE must be provided. The MAGE might need to consist of high, removable platforms and scaffolds, or a 19' ground clearance (half Solar Array span) turnover fixture with compatible scaffolds and platforms. This MAGE might be required either inside or outside the chamber depending on the assembly approach. These MAGE requirements may be relaxed by some combination of support system involvement in the assembly operation either inside or outside the chamber.

The requirement on the support system to accommodate spacecraft of varied sizes and launch vehicle interfaces implies that a spacecraft adapter structure will be necessary in cases where spacecraft have trunnion type hardpoints. The trunnion supported spacecraft of 14 ft. typical diameter, (designed for shuttle launch only) has a significantly different launch interface from that of the DSCS III. This requirement, however, is common to, and affects equally, both the gimballed stand and suspension support approach and will not affect their comparison.

3-6 SATELLITE SUPPORT SYSTEMS.

Six candidate support systems presented in Figures 3-10 to 3-15 were evaluated for use in the SXTF. They are the result of the screening of several satellite support system concepts. Concepts 1 to 3 utilize the suspension approach, while concepts 4 to 6 utilize the gimballed rigid fixture approach.

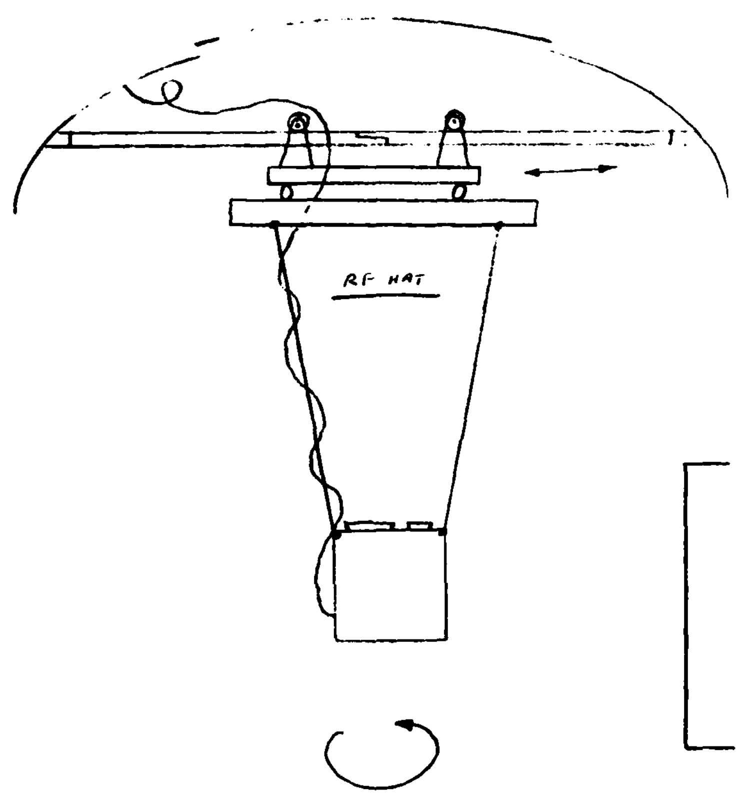
Concept 1, shown in Figure 3-8, is mechanically the simplest of all the concepts shown. It consists of eight dielectric lines secured to hard points



FEATURES

PRO	CON
- VERY SIMPLE	- NO CAPABILITY OF REMOTE S/C TRANSLATION OR ROTATION
- LOW COST	- HIGH REPOSITIONING TIME COSTS
- RELIABLE	- NO CAPABILITY OF REPOINTING
- LOW MAINTENANCE	
- ALL DIELECTRIC MATERIALS (EX. AT S/C & WALLS)	

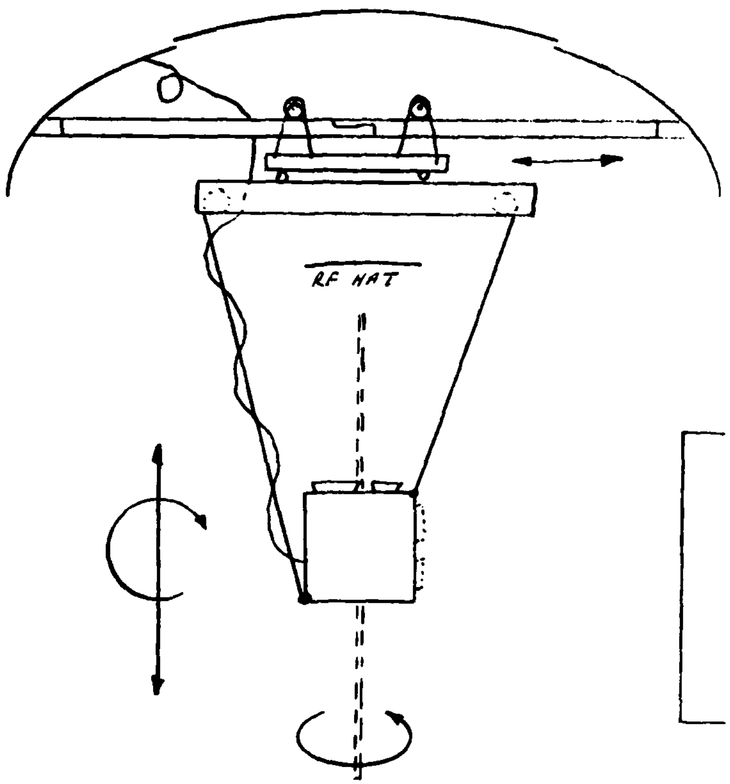
Figure 3-10. Support system #1.



FEATURES

PRO	CON
- REMOTE S/C HORIZONTAL TRANSLATION CAPABILITY	- NO REMOTE CAPABILITY OF S/C VERTICAL TRANSLATION
- REMOTE S/C ROTATIONAL CAPABILITY ABOUT VERTICAL AXIS	- MINIMUM REMOTE RF REPOINTING
- DIELECTRIC MATERIAL ONLY IN VICINITY OF S/C	- NOT AS SIMPLE AS SYSTEM #1
- MORE RELIABLE THAN SYSTEM #3	- INTRODUCTION OF SOME METAL ELEMENTS INTO CHAMBER (ONLY AT TOP)
- MAINTENANCE REQUIRED	- REQUIRES OVERHEAD SUPPORT STRUCTURE/RAILS
	- LESS RELIABLE THAN SYSTEM #1
	- SIGNIFICANTLY MORE COSTLY THAN SYSTEM #1 (LESS COSTLY THAN #3)

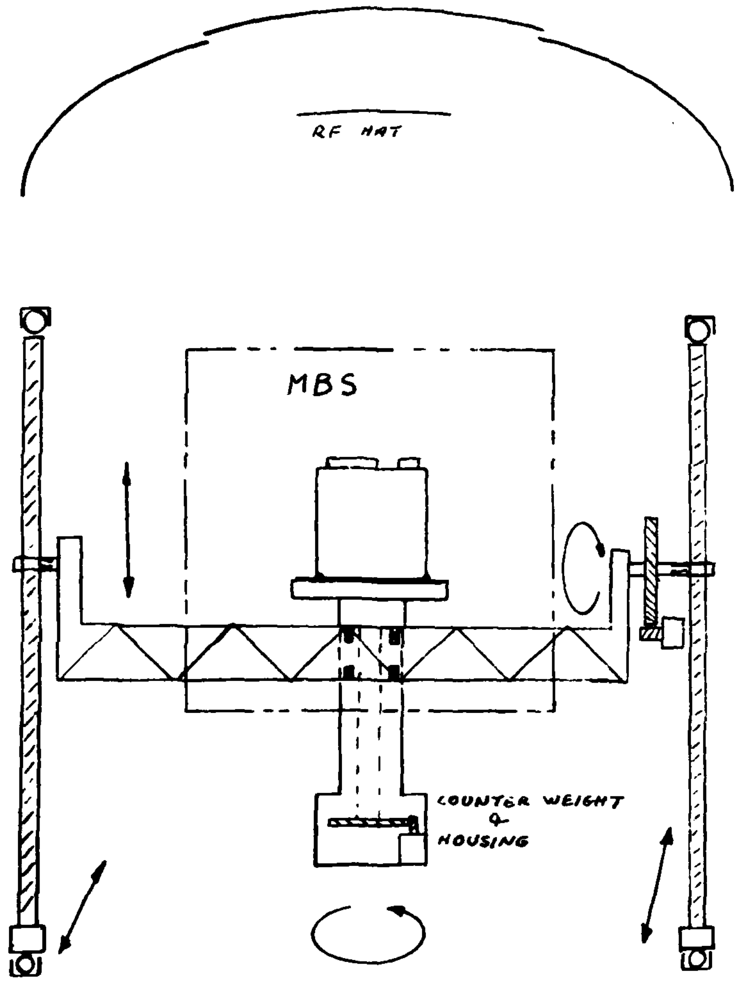
Figure 3-11. Support system #2.



FEATURES

PRO	CON
- REMOTE VERTICAL TRANSLATION CAPABILITY	- REQUIRES OVERHEAD SUPPORT STRUCTURE/RAILS
- REMOTE HORIZONTAL TRANSLATION CAPABILITY	- NOT AS SIMPLE AS SYSTEM #1 OR #2
- REMOTE S/C ROTATIONAL CAPABILITY ABOUT VERTICAL AXES	- LESS RELIABLE THAN SYSTEM #1 OR #2
- DIELECTRIC MATERIAL ONLY IN VICINITY OF S/C	- INTRODUCTION OF SOME METAL ELEMENTS INTO CHAMBER (ONLY AT TOP)
- MODERATE MAINTENANCE	- USE OF MORE MOTORS IN VACUUM SIGNIFICANTLY AFFECTS RELIABILITY
- REMOTE RF REPOINTING CAPABILITY	- LIKELY COSTLIEST OF SUSPENSION SYSTEMS

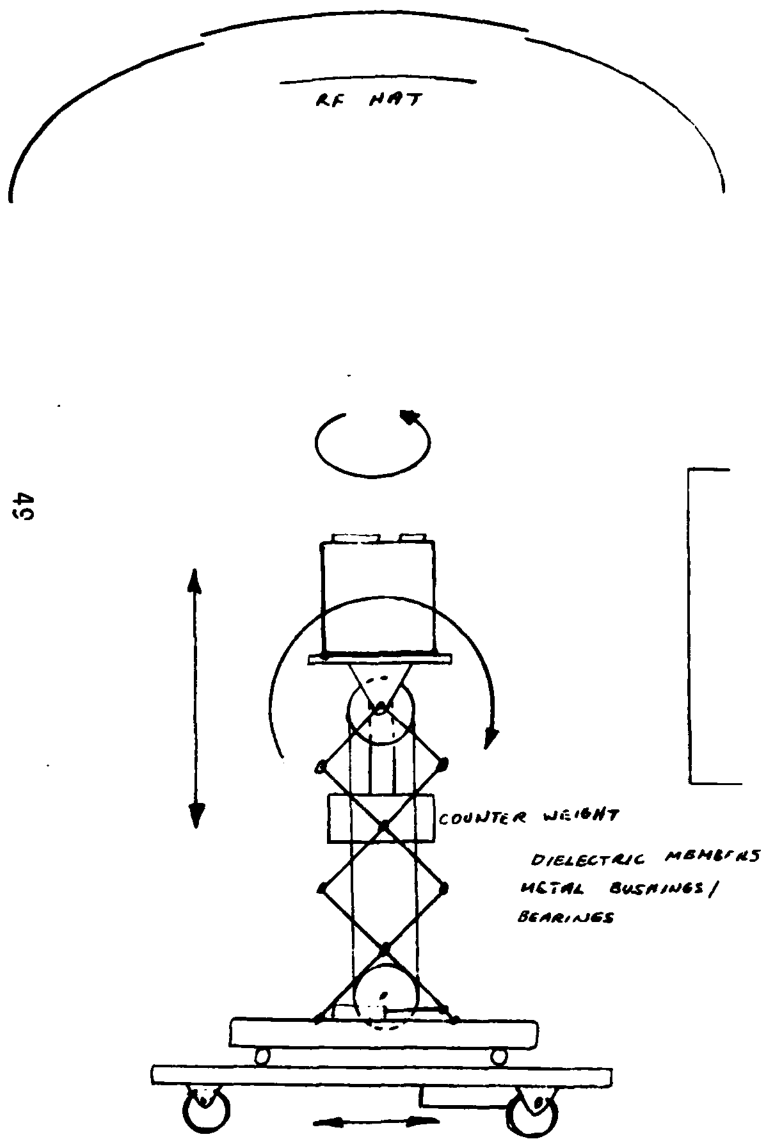
Figure 3-12. Support system #3.



FEATURES

PRO	CON
- REMOTE S/C TRANSLATION CAPABILITY (VERTICAL & HORIZONTAL)	- INTRODUCTION OF METAL IN VICINITY OF S/C
- REMOTE S/C ROTATION CAPABILITY (2 AXES)	- GREATEST MAINTENANCE PROBLEM
- REMOTE RF REPOINTING CAPABILITY	- MOST COMPLEX
- BEST CAPABILITY OF MECHANICAL SUPPORT SYSTEMS FOR HANDLING LARGE S/C	- MOST COSTLY
	- LEAST RELIABLE

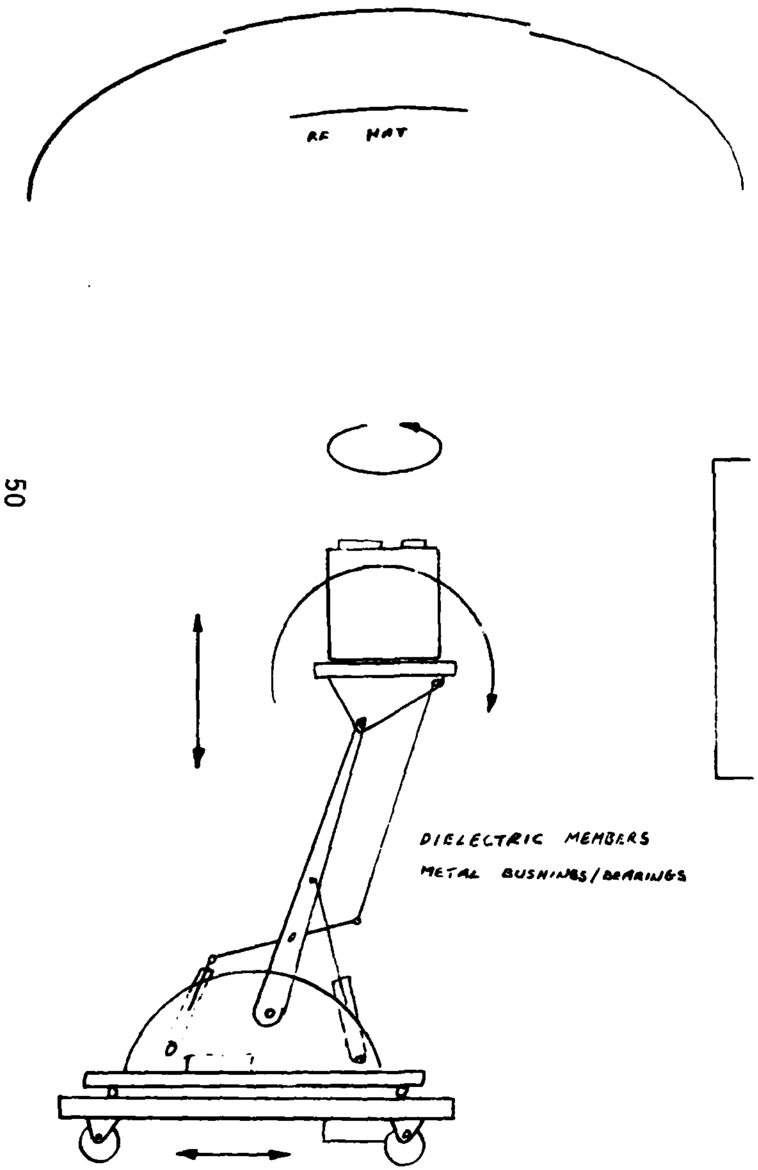
Figure 3-13. Support system #4.



FEATURES

PRO	CON
- REMOTE S/C TRANSLATION CAPABILITY (2 DIRECTIONS)	- INTRODUCTION OF METAL IN VICINITY OF S/C
- REMOTE S/C ROTATION CAPABILITY (2 AXES)	- SIGNIFICANT MAINTENANCE
- REMOTE RF REPOINTING CAPABILITY	- MORE COMPLEX THAN SUSPENSION SYSTEMS THAN SYSTEM #6
- MODERATE COMPLEXITY (OF MECH. SUPPORT SYSTEMS)	- 2ND MOST COSTLY SYSTEM
- MODERATE COST FOR MECHANICAL SUPPORT SYSTEMS	- LESS RELIABLE THAN SUSPENSION SYSTEMS #1 AND #2
- MODERATE RELIABILITY FOR MECHANICAL SUPPORT SYSTEMS	

Figure 3-14. Support system loading #5.



FEATURES

PRO	CON
- REMOTE S/C TRANSLATION CAPABILITY (2 DIRECTIONS)	- INTRODUCTION OF METAL IN VICINITY OF S/C
- REMOTE S/C ROTATION CAPABILITY (2 AXES)	- SIGNIFICANT MAINTENANCE
- REMOTE RF REPOINTING CAPABILITY	- MORE COMPLEX THAN SUSPENSION SYSTEMS
- SIMPLEST OF MECHANICAL SUPPORT SYSTEMS	- COSTLY COMPARED TO SUSPENSION SYSTEMS
- LEAST COSTLY OF MECHANICAL SUPPORT SYSTEMS	- LESS RELIABLE THAN SUSPENSION SYSTEMS #1 AND #2
- MOST RELIABLE OF MECHANICAL SUPPORT SYSTEMS	- BASIC APPROACH IS UNWIELDY - ESPECIALLY FOR LARGE S/C
	- REMOTE TRANSLATION SEVERELY LIMITED IN TOTAL DISTANCE
	- SEVERE INTERACTION BETWEEN DEGREES OF FREEDOM AND STRUCTURAL LOADING.

Figure 3-15. Support system loading #6.

on the chamber ceiling, walls and floor. Limited adjustment in the lines is provided by turnbuckles located close to the chamber hardpoints to minimize electrical interference. Mounting to the vehicle is by swivel-like fittings bolted to the longeron hardpoints. Reconfiguration of the vehicle with this system is completely manual, thereby necessitating entrance to the chamber for S/C repositioning and contributing to significant operating costs in long term use. With each new configuration, lines of different lengths are required. The system has very good thermal and electrical isolation characteristics, and shadowing is minimal. Its compatibility with the deployed solar arrays and solar array support system is good, with the exception that at some oblique angles of the vehicle, with respect to the MBS, there may be interference of the supported solar arrays with the suspension lines.

The remotely controlled translation along the horizontal MBS axis and rotation about the vertical, as shown in Figure 3-9, are the outstanding features of Concept 2. The superstructure in the chamber consists of a double rail system which permits easy removal during vehicle entry and exit operations. A motorized hanging carriage is supported from these rails. The carriage, in turn, supports a rotating stepper motor driven strongback which has at least four hardpoints to which dielectric vehicle supporting lines are attached. The dielectric lines attach to the vehicle by swivel fittings at each of the eight hard mounting points on the vehicle, as in Concept 1. The strongback design will be such that the line angled to the satellite will provide the necessary stability. One or two of the lines could be used to provide support or alignment for the ground power and fiber optics umbilical.

The advantages of Concept 2 include low operating cost due to a significant reduction in handling for reconfiguration, and good thermal and electrical

isolation. The system's accurate repositioning capability has the associated disadvantage of lessened reliability due to the presence of redundant electric motors and a generally more complex mechanism. This system would also accommodate an RF coupler system located or possibly attached to the bottom of the strongback.

As in Concept 1, this concept also shows the test orientation exposing the center body only without the solar arrays deployed. Although this is a prime test orientation, a manual realignment of the vehicle with the antenna farm facing the chamber wall or MBS would allow deployment of the solar arrays. Concept 2 would allow this configuration including an oblique orientation, as illustrated in Figure 3-6. The deployed solar array could, however, experience interference with the dielectric suspension lines if certain configurations are chosen.

The features of Concept 3, shown in Figure 3-12, are identical to those of Concept 2 except that additional remote repositioning capability is incorporated by means of hoists on the rotating strongback. The dielectric suspension lines would attach to the vehicle by swivel fittings at opposite corners of the vehicle allowing horizontal vehicle rotation. Vertical translation is also possible by operating the hoists in unison.

Concept 3 has the advantage of greatly reduced lifetime operating cost because of its total remote reconfiguration capability and its vertical translation capability which permits partial solar array and center body exposure and adaptability to accommodate vehicles of various sizes. It is somewhat less reliable than Concept 2 because of the additional hoists and system complexity. Concepts 3 and 2 share the complexities of the superstructure and vehicle entry and exit operations. These complexities take the form of possible

interference between the support system and the twenty foot diameter chamber lid and the degree of additional handling inside and outside of the chamber due to the possible blockage of the lid opening by the support system during the satellite assembly.

The structure of Concept 4, shown in Figure 3-13, like those of the other two gimballed rigid support fixtures considered, when properly designed would consist primarily of dielectric materials. A rotating vehicle platform also of dielectric material would interface between the vehicle and bridge structure. Several hardpoints on this platform would provide for mating to the hard mounting points of the test vehicle. The vehicle platform would be supported by a dielectric bridge structured platform such that it may rotate with respect to the bridge. The bridge structure would be designed with dielectric truss members to minimize the dielectric cross-sectioned area and thus reduce shadowing and electrostatic charging effects during the test while providing sufficient support for vehicle weights considered.

Rotational capability would be provided and loading would be carried by two large bearings set into the bridge platform. A lightweight dielectric housing supported by the bridge would be used to align the shaft, gears, motor, and counterweights of the vehicle platform. Rotation about the horizontal axis is provided by a gear and motor assembly located at one end of the bridge platform. The entire assembly of platforms, vehicle, and counterweights would be translated vertically by four actuators providing redundancy on each end. These actuators can also be motor driven along two tracks which are parallel to the MBS horizontal axis, thus providing remote MBS to vehicle distance control. For operation in the thermal vacuum environment, motors, gears, and actuators fitted with strip heaters would be required. These have the

effect of reducing the electrical isolation between the vehicle and chamber. However, electrical isolation is maximized by locating metallic gears, motors, and actuators close to the chamber walls and away from the vehicle.

Remote repositioning capability is fully realized in this system which has the capability of two rotations and two translations. This system also provides high RF pointing accuracy, which is typical of gimballed rigid systems. Vehicle handling capabilities are greatly enhanced in this system concept for the following reasons:

a) The system does not encroach into the hoisting zone of the vehicle;

b) Large vehicles can be accommodated both spatially and by weight.

c) The rigid fixture provides the best opportunity for assembling the solar arrays and vehicle inside the chamber.

The disadvantage of this concept is that is has the highest initial design and fabrication cost resulting from its complexity and high relative engineering design effort. It also has the least reliability due to the large number of electrical and mechanically moving parts. This concept also shares with the other rigid support systems the significant thermal presence of the structure, the reduced electrical isolation due to metallic elements relatively close to the vehicle, and the shadowing of the vehicle/support system interface surface from x-ray exposure. It also presents a very large dielectric surface which will become charged in an electron plasma environment and which may interfer with the vehicle's system response during the test.

The vehicle/support interface platform of Concept 5, shown in Figure 3-14 is similar to that of the previous concept except that the dielectric shaft and counterweight are attached directly to the vehicle platform. A gusset interfaces the vehicle platform with the dielectric membered scissor-jack.

Remote positioning of the test object about the horizontal chamber axis is accomplished by a dielectric wheel and belt drive attached to the vehicle platform gusset, which is driven by a similar mechanism located on the bottom platform. The scissor-jack consists of dielectric structural members with steel bearing joints. It is supported by a rotating platform containing motors and driven wheels, which elevates the superstructure, and rotates the platform about the vertical, and drives the vehicle horizontal rotation position. The rotating platform is in turn supported by a tracked electric motor driven base, which provides translation along the horizontal MBS axis. An alternative to the belt drive system might consist of a system of dielectric levers and linkages.

The outstanding advantage of Concept 5 is its smaller cross sectional area than the structure in Concept 4 while still providing total remote reconfiguration. This reduces the electrostatic charging interference with the test vehicle and the total weight to the support structure. With fewer motors, it should be more reliable than the bridge concept. The major disadvantages include its structural softness along the vertical axis and its reduced electrical isolation due to metal bearings in its linkages. To alleviate this, guidelines may be required from the bottom platform to the vehicle to provide extra stability. The electrical isolation will also be degraded by a possible requirement for heater strips and their power leads around the bearing and motors in order to ensure proper operation in the thermal vacuum environment. In addition, the concept of a belt driven system to achieve a horizontal rotation capability is only effective for one vertical height and would require a re-work for each different test elevation.

Concept 6 consists of a dielectric vehicle interface platform supported and repositioned by two sets of dielectric linkages. The linkages are supported by a rotating platform similar to that of Concept 5. Actuators on this platform control the linkages which provide horizontal rotation and vertical translation to the vehicle platform. Rotation about the vertical axis and horizontal translation are provided by the same approach described for the Concept 5.

The major advantage of this system is its relative simplicity as a gimballed fixture. However, it has a major disadvantage from a structural loading viewpoint, since more severe loads must be carried by its fewer members, and the system footprint will be the greatest of the gimballed systems due to off-centered loading. Although remote reconfiguration is possible in two translation and two rotations, all these degrees of freedom are coupled by the linkages and the vehicles' center of gravity offset.

3-7 CURRENT RECOMMENDATIONS.

Mechanically, the simplest concept presented is Concept 1. It has, however the outstanding disadvantage of being incapable of remote repositioning. Although it is a highly reliable proven (Huron King) concept, its high operating costs, resulting from the multiple repressurizing of the chamber and manually reconfiguring the satellite, eliminates it as a candidate for the SXTF.

Concept 6 like 1 appears quite simple mechanically. However, the offset center of gravity generates very large loads in several areas of the structure and requires that the base have some unique stability characteristics including a very large footprint. Because of coupling among the degrees of

freedom, many motions become limited in scope. For example, vertical translation of the satellite will require some horizontal rotation, and translation of the base toward the MBS, if satellite distance from that source is to be maintained. Therefore Concept 6 is among the less attractive candidates.

The two gimballed rigid support concepts 4 and 5, though mechanically dissimilar, are quite competitive by virtue of their key features. The envelope of the mechanical operation of Concept 5 is relatively small, and the less structural surface area, makes this concept more attractive than Concept 4 where charging and thermal coupling effects are concerned. However, a mechanical disadvantage of Concept 5 is its belt drive or linkage system which controls horizontal rotation. It does not have the flexibility of the scissor-jack concept and providing them would significantly complicate the system. The present configuration of Concept 5 requires chamber repressurization for linkage adjustment if vertical translation is needed, for example, for x-ray exposure of the solar arrays following center body exposure.

The presence of more motors and actuators in Concept 4 reduces its reliability. However, motors are quite reliable devices in themselves. Therefore, Concept 4, when viewed in the context of its operational life, has greater value due to its key capabilities of total remote control and adaptability to larger vehicles. Its structural rigidity could also facilitate in chamber solar array installation. Therefore, Concept 4 is preferred over Concept 5 as the most viable of the gimballed, rigid supports for the SXTF.

The mechanical aspects of Concepts 2 and 3 are very similar. Their major advantages over the gimballed supports are their ability to remotely reconfigure the vehicle with sufficient accuracy to satisfy RF requirements, and yet be well isolated thermally and electrically. Their extremely small

mass in the area adjacent to the satellite and the MBS are particularly attractive from a shadowing and charging viewpoint. Both concepts are compatible with the deployed solar arrays orientation, although some interference can be experienced at oblique angles.

As discussed above, the negligible loss of reliability when viewed against the long-term gain in cost effectiveness, is the basis for selecting Concept 3 as the prime candidate for the SXTF. With the addition of hoisting motors, Concept 3 becomes considerably more maneuverable than Concept 2, and therefore probably more cost effective in the long term.

SECTION 4

SURVIVABILITY REQUIREMENTS

4-1 SXTF TEST MODES.

The SXTF is essentially a low-level x-ray simulator and threat-level ECEMP simulator which can be used in a number of modes to obtain system response data directly applicable to low-level x-ray threats. The data from such testing can then be used to extrapolate to higher level x-ray threats. In addition, the SXTF can be used for repeated exposures such that many active modes of the satellite may be investigated.

The various test objectives and simulator modes are given in Table 4-1 along with appropriate comments. All of the test objectives could be met with a sequence of tests on an active satellite involving AGE and EMP instrumentation. Spacecraft charging effects (ECEMP) resulting from natural and nuclear electron environments could also be addressed in the same test sequence.

A low level upset and recovery test of the satellite electronics can be performed in order to evaluate upset and recovery of the satellie including any potential system level interactions. This type of test could be performed on a full-up system including deployed solar arrays. This configuration would allow for combined TREE/SGEMP test. The primary SGEMP effects for this test configuration would be the external SGEMP response of the satellite. A higher level TREE test of the electronics can be performed by exposing only the satellite centerbody. In this configuration, however, the correct external SGEMP effects would not be present. The flash x-ray exposure of the instrumented centerbody is designed to catch TREE malfunction modes which might be displayed at

Table 4-1. SXTF test modes.

TEST OBJECTIVE	SIMULATOR SOURCE	PURPOSE	CONFIGURATION	COMMENTS
Low-Level Upset				
• Combined Effects	MBS/PRS	System level TREE/SGEMP Upset test.	Centerbody with Solar Array deployed	Evaluate system level interactions
• TREE	MBS	Higher level TREE test of electronics in center body plus cavity/cable SGEMP.	Centerbody only	
Shielding Effectiveness				
• Faraday Cage and penetrations	PRS	Measure Faraday cage and penetration shielding effectiveness.	Centerbody with Solar Array deployed	SGEMP response extrapolat to higher levels required
• Cable shields	MBS	Measure coupling of cavity fields to cables.	Centerbody only	Assumes Faraday Cage shielding is good because external leakage into cavity not reproduced correctly without solar arrays
Threat level ECEMP and ECEMP/SGEMP synergism	EB/MBS/PRS	Threat level electron irradiation of satellite with possibility of simultaneous MBS/PRS exposures.	Centerbody only	

the system level, as opposed to the box level. The surprise system-level malfunction modes would tend to be associated with upset recovery, as opposed to burnout, and would therefore require AGE instrumentation support.

The shielding effectiveness tests using the PRS/MBS sources can be used to evaluate Faraday Cage penetrations from the exterior to the interior of the satellite. SGEMP response data at low levels can be obtained in MBS exposures of the center body for use in extrapolating the SGEMP response to threat level. The cable and cavity field response should be essentially linear up to threat levels, and data obtained at lower levels could be directly extrapolated. This type of data could be used to compare with interface current design levels and current injection test data on the box interface.

The ECEMP response of the center body would be useful for confirming worst-case design current levels, as well as uncovering any unexpected satellite electronic response. Both high- and low-energy electrons would be required.

4.2 CANDIDATE SATELLITE TEST ORIENTATIONS.

The principal external configuration features of the DSCS III satellite are shown in Figure 4-1. This satellite is representative of more advanced military communications satellite payload systems. The spacecraft measures 458 inches tip-to-tip, with the solar arrays deployed, and is configured with a 110 inch long, 76 inch wide and 77 inch deep center body, and has a dry weight (weight without hydrazine propellant)in excess of 1600 pounds. The principal DSCS III structural elements are shown in Figure 4-2.

The SGEMP hardening design utilizes the spacecraft structural elements as the first level of RF shielding. The S/C configuration is designed to house the component electronics in two structural sections---the north panel housing and the south panel housing. The form factor of this structure totally encloses electronics and harnesses forming a Faraday Cage shielding enclosure. Electrical field components from spacecraft charging products and EMP will be highly attenuated due to the highly reflective shielding

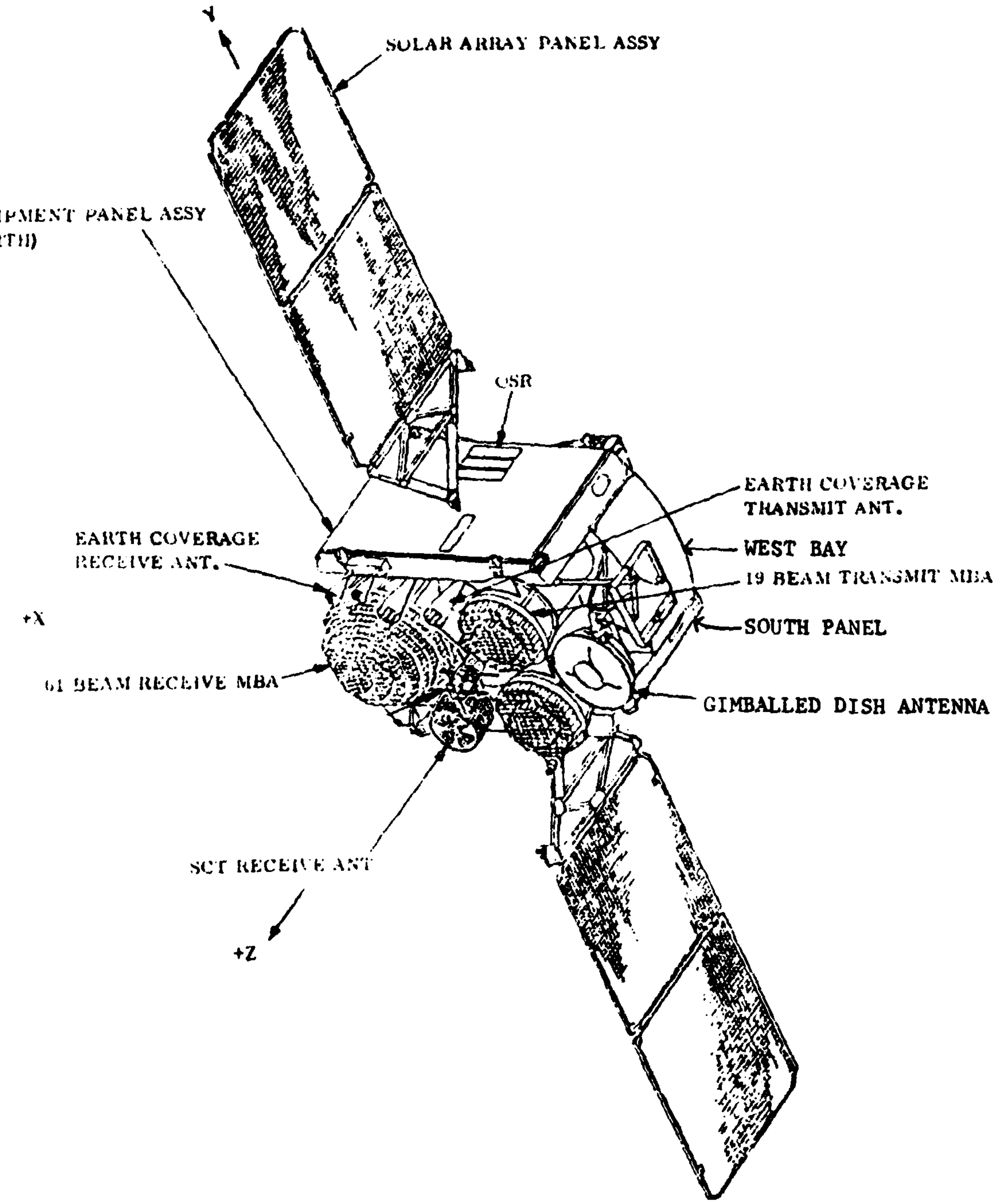


Figure 4-1. External configuration of the DSCS III satellite.

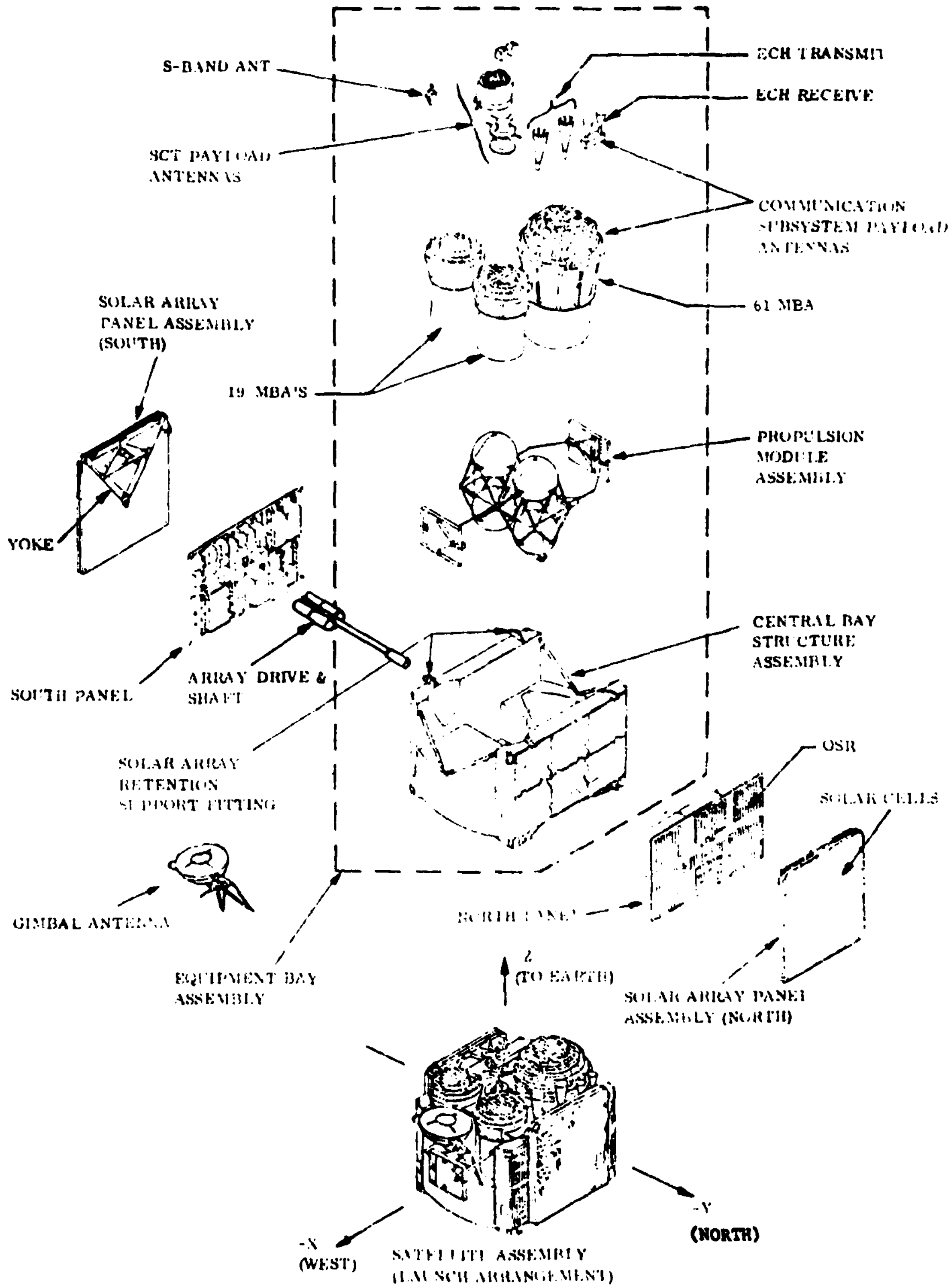


Figure 4-2. DSCS III structural elements.

characteristics of the structure.

There are four shielded enclosures - i.e., south, north, east and west bays. The south bay contains the main housekeeping control and power components, the north bay, the major communications units, the east bay the antenna controls and the west bay the connections among the bays. The south and north bays resemble compartmented arrays closed in the rear. Components are mounted on panels which are secured to the bays, thus forming sealed compartments.

Based on the studies of the DSCS III and STARSAT vehicles and the SXTF test modes, six candidate test orientations are suggested for evaluation in the SXTF environment. The spacecraft orientations relative to the radiation sources for these tests are shown in Figure 4-3.

NORTH PANEL EXPOSURE

The orientation shown in Figure 4-3A is chosen to obtain a high dose on the communications payload on the North Panel. This panel would therefore be placed as close to the radiation source as possible. SGEMP response data relative to cavity fields, cavity field coupling to cables and cable direct drive can be obtained for use in extrapolating to higher levels.

SOUTH PANEL EXPOSURE

The orientation shown in Figure 4-3B is similar to configuration A except that a high dose on the housekeeping (command, telemetry, power and station keeping) electronics is obtained. An option for this configuration is the addition of the solar array panels either folded next to the South Panel or folded adjacent to the Centerbody, e.g., with jumper cables. Exposing the solar arrays is important because the solar array response is the major stimulus to the satellite's primary power subsystem. The placement of the solar arrays relative to the centerbody would be

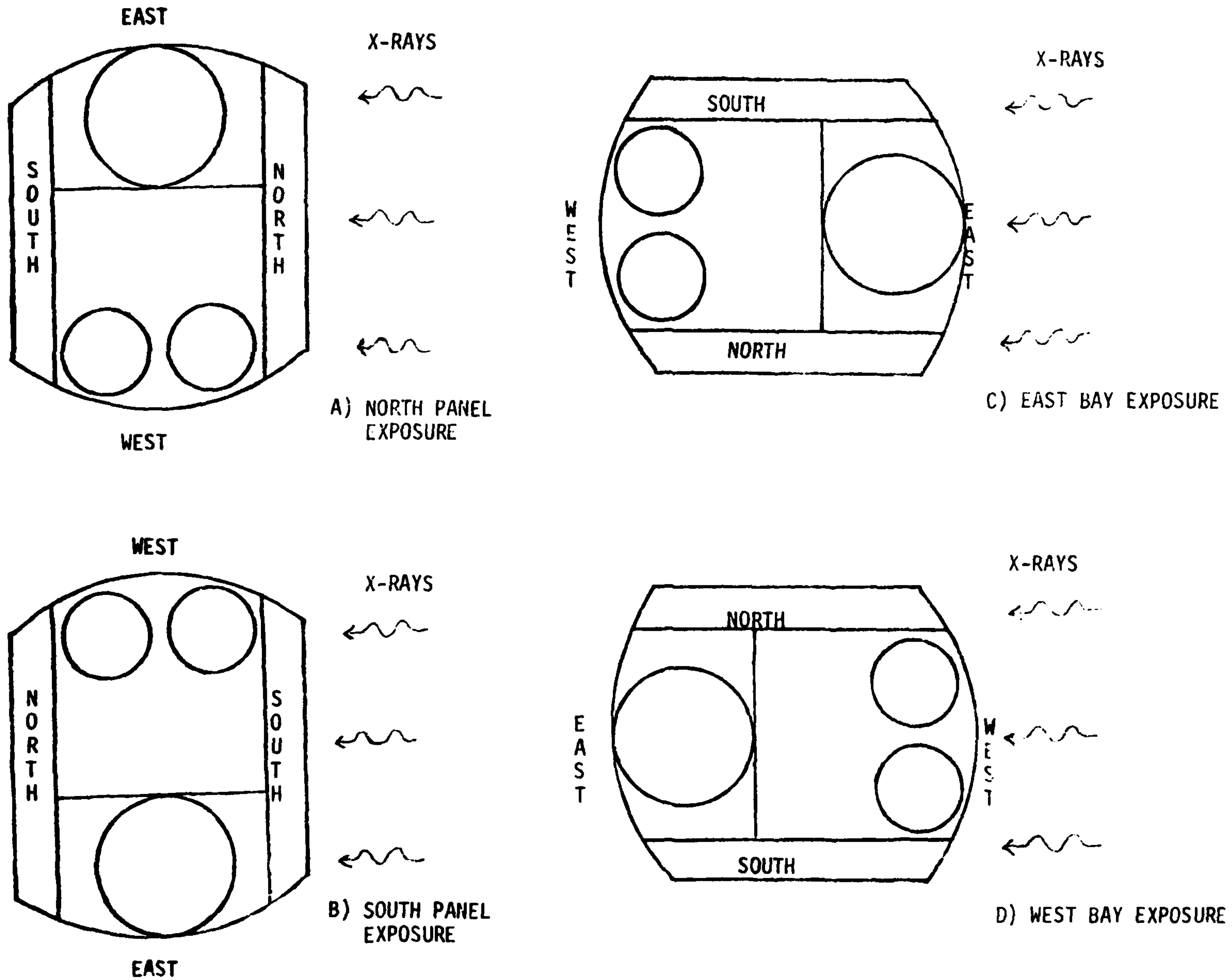


Figure 4-3. Spacecraft orientations in SXTF for various test options.

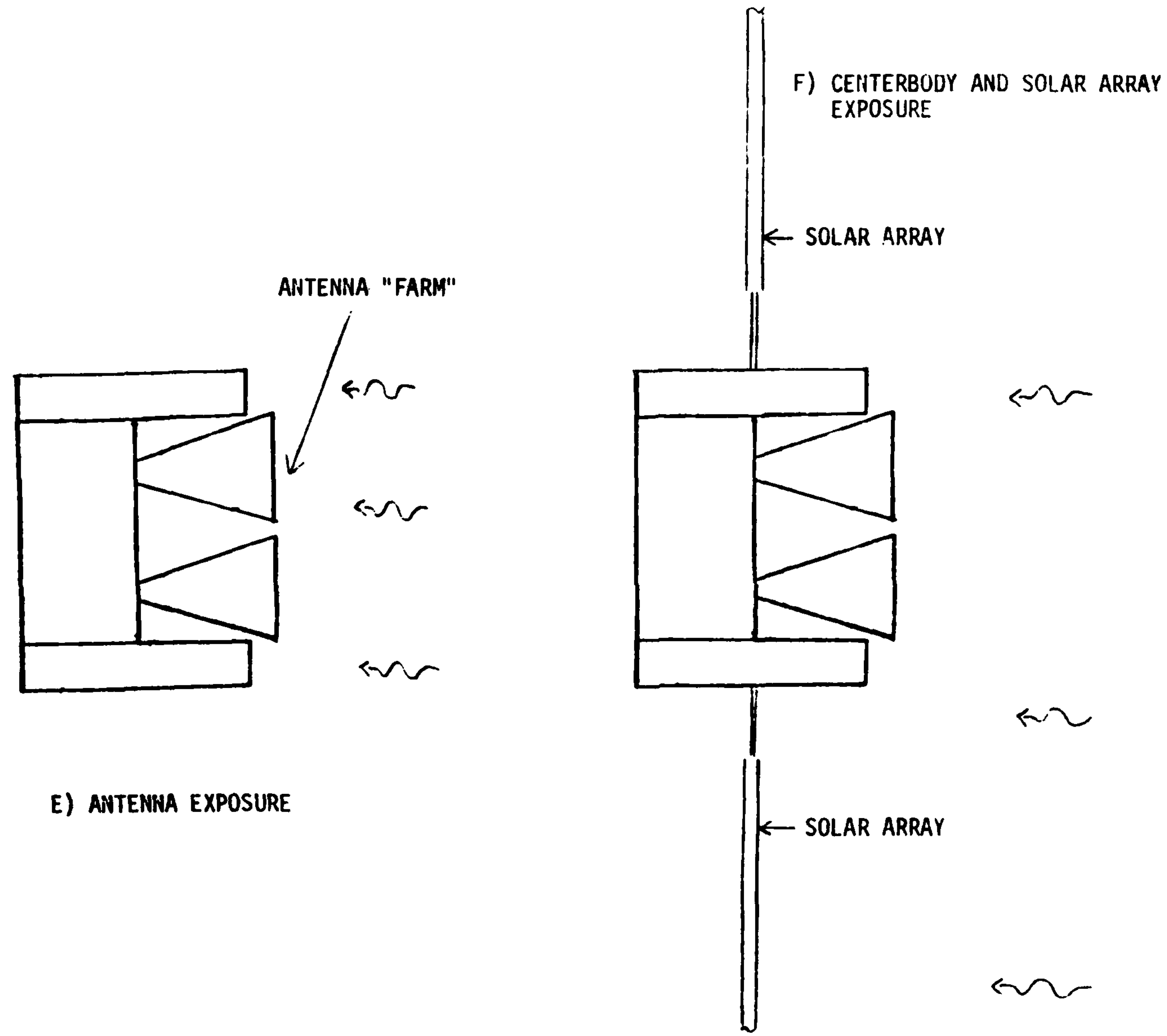


Figure 4-3. Spacecraft orientations in SXTF for various test options (Cont'd).

determined by the amount of X-ray shielding of the South Panel Electronics caused by the arrays. Exposures of the North and South panel are necessary rather than just one panel, because of the x-ray shielding afforded one panel by the other panel and the rest of the spacecraft centerbody. Tilting the satellite at an angle would not significantly help in this regard because the centerbody structure, primarily the propulsion tanks, still shields most of the panel.

EAST BAY EXPOSURE

The purpose of the East Bay Exposure, shown in Figure 4-3C,is to perform a TREE test on the jammer location electronics (JLE) and multi-beam antenna (MBA) electronics located in this bay. Cavity IEMP fields and coupling are the major drivers for the electronics in this cavity because of the presence of cables. For practical thermal control and packaging reasons on the MBA electronics, these cables are unshielded.

WEST BAY EXPOSURE

The West Bay Exposure, shown in Figure 4-3D, excites the largest cavity in the spacecraft. Although there are no electronics in this cavity, data on the IEMP coupling to the long power, command and telemetry system cables that connect the North Panel Electronics to the South Panel Electronics are needed. Both the East and West Bay Exposures are necessary because of the x-ray shielding afforded by the centerbody structure, primarily the propulsion tanks.

ANTENNA EXPOSURE

The exposure shown in Figure 4-3E, excites the satellite antenna "farm" located on the top of the vehicle. This orientation maximizes the X-ray shielding caused by other parts of the satellite structure. The antenna "farm" would be positioned as close as possible to the radiation source and the SGEMP response would be extrapolated to higher levels.

CENTERBODY AND SOLAR ARRAY EXPOSURE

In the satellite orientation shown in Figure 4-3F both the centerbody and one of the solar arrays are exposed to the X-rays. This exposure maximizes the external structure SGEMP response of the satellite, while providing a lower level simultaneous exposure of the satellite electronics. This allows for a low level combined effects test to investigate any system level synergistic interactions. This orientation may be at 90^{0} or oblique to the radiation source in order to maximize system response and minimize source shielding.

4-3 SURVIVABILITY REQUIREMENTS

Typical survivability test requirements for the test spacecraft are derived from the Air Force/Space Division system survivability specifications by the process shown in Figure 4-4. A system level Functional Analysis assesses each spacecraft (S/C) function to define the system effect to transient upsets in the component elements. The limits of acceptable response are used to apportion transient performance to formulate system and component design requirements. These requirements are used as evaluation criteria for the system TREE and SGEMP/EMP Analyses. The survivability qualification tests verify the performance of the spacecraft subsystems against these same criteria. Unacceptable component performance include burnout, degradation and some types of transient response (both state change and recovery). A summary of how these different types of requirements would be verified using the SXTF is given in Table 4-2. Burnout and degradation effects would be evaluated by post shot telemetry checks and full functional tests of the spacecraft. Transient response requirements would be evaluated by either spacecraft telemetry or direct measurements.

The measurements to verify the component box transient recovery requirements include both the spacecraft telemetry and direct SGEMP and TREES measurements. The spacecraft telemetry contains about 300 measurement points

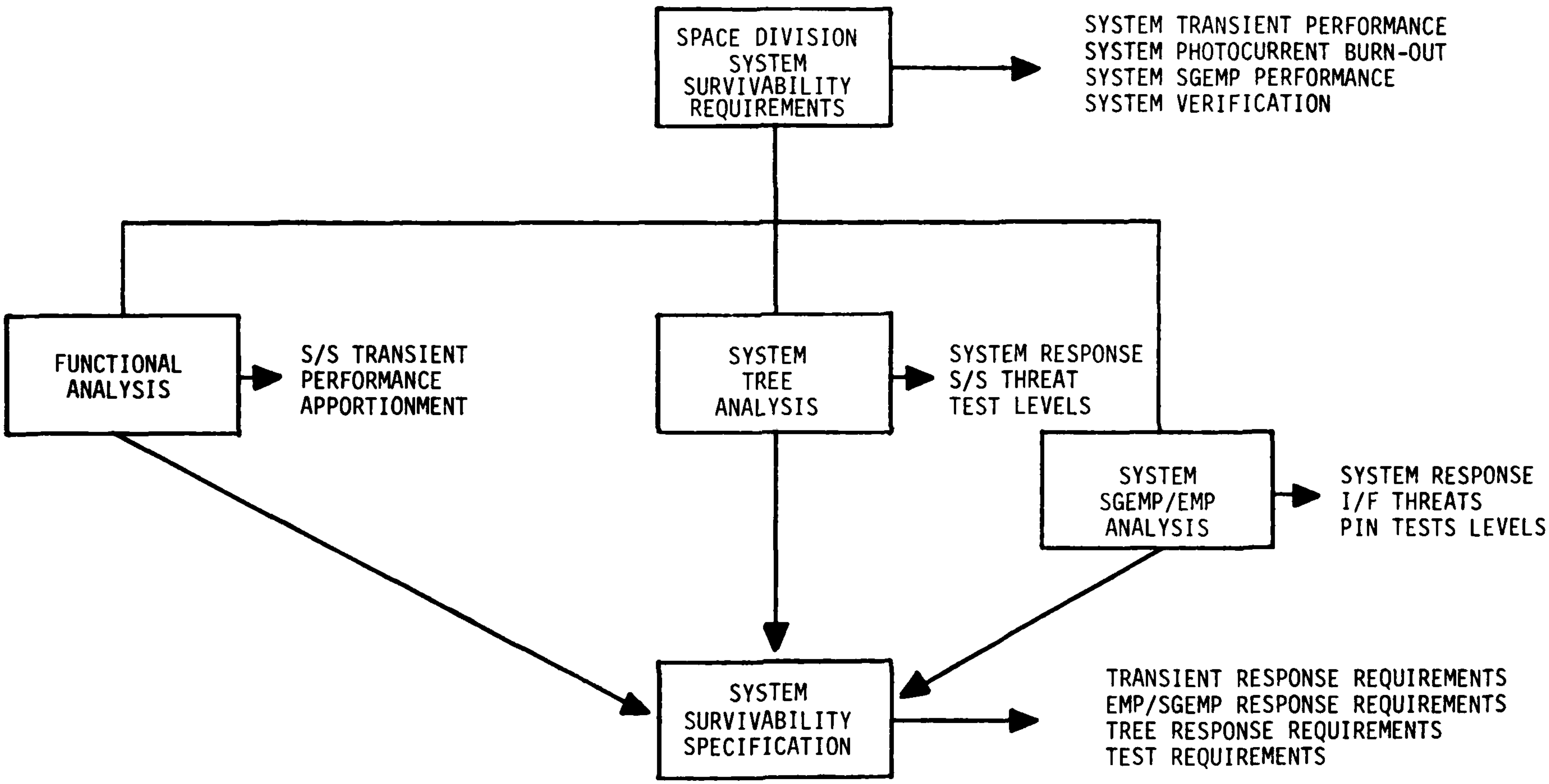


Figure 4-4. Derivation of system survivability requirements.

Table 4-2. Electronics measurement summary.

EFFECT	MEASUREMENT TECHNIQUE
• BURNOUT	• POST SHOT TLM, FACTORY TEST
• DEGRADATION	• POST SHOT TLM, FACTORY TEST
• TRANSIENT RESPONSE:	
- STATE CHANGE	• IN-SITU, POST SHOT TLM
- RECOVERY	• IN-SITU, DIRECT MEASUREMENT VIA PROBE OR OF A TLM POINT.
	• POST SHOT PERFORMANCE

of analog, bilevel and serial digital data. This data would be transmitted over an X-Band air link during an SXTF Test. The spacecraft will also be commanded via this link.

The direct measurements would be made using sensors added to the spacecraft prior to the test and utilizing fiber optic links to transmit the data. To determine the requirements for direct measurements, it is necessary to understand the operation of the spacecraft electronics and their transient recovery requirements.

The following discussions present a brief functional description of each of the electronic subsystems on a baseline test vehicle, such as the DSCS III satellite. The key nuclear survivability phenomologies associated with each component box from a functional subsystem standpoint are presented together with a description of the survivability design requirements and the measurements required to verify them. Table 4-3 lists the components for each of the five major subsystems, which are:

1. Power Subsystem
2. Single Channel Transponder (SCT) Subsystem
3. Communications Subsystem
   - Beam Forming Network (BFN)
   - Jammer Location Electronics (JLE)
4. Telemetry, Tracking and Command Subsystem
5. Attitude Control Subsystem (ACS)

POWER SUBSYSTEM

The Electrical Power and Distribution Subsystem (EPDS) provides for the conversion of solar energy to electrical power, the regulation and distribution of power to the spacecraft subsystems (load) and the battery storage of a portion of the electrical energy for subsequent use during

Table 4-3. DSCS III electronics required for S/C level test.

POWER S/S	COMM S/S	SCT S/S	ACS	TT&C S/S
SOLAR PANEL	61 BFN	SHF RCVR	SUN SENSOR	S-BAND XPNDR
PRU	19 BFN	UHF RCVR	EARTH SENSOR	CTU
SPC	RF SWITCH	DIGITAL PROCR	ACE	RTU
NPC	RF DETECTOR	FREQ. SYN.	REACTION WHEEL	SHF/S
SHUNT DISSIPATOR	JLE DGTL CNTRLR	SCT PWR CNTLR		S/IF
BATTERY	LNA	UHF XMTR		BEACON
	RLM	KI-35		KIR-23
	TDAL			KI-24
	MIXER			KGX-28
	FREQ. STD.			
	FREQ. GEN.			
	DRIVER AMP			
	TWTA			
	TLS			
	C/T IFU			

periods when the solar array capability is inadequate.

The EPDS consists of the following elements associated primarily with the generation, regulation and storage of electrical energy:

a. Four Solar Array Panels (SA)

b. Two Shunt Dissipators (SD)

c. One Power Regulation Unit (PRU)

d. Three Nickel Cadmium Batteries

Thus, the EPDS consists of the equipment necessary to provide:

a. Generation and storage of electrical power (energy).

b. Distribution of conditioned power to users.

c. Distribution of electrical signals (excluding RF signals) between equipment and the protection of such interconnecting cabling from the effects of electromagnetic interference.

d. Protection of the power bus from detrimental transient loads, shorts and the effects of load sequencing and removal.

e. Telemetry signals sufficient to monitor the EPDS status and to diagnose EPDS faults.

f. Execution of commands received from the Telemetry, Tracking and Command Subsystem (TTCS).

The key nuclear survivability concerns with this subsystem are:

1. Transient Performance

   a. Solar Array photovoltaic response

   b. Shunt Dissipator solar array snubber response

   c. PRU control loop response

   d. PRU Failure Detector mode change

2. Photon Direct Drive SGEMP

   a. PRU interface threat

3. External SGEMP

a. Structural current response with SADAPTA as main spacecraft point of entry(POE).

The specific survivability transient recovery and verification requirements for this subsystem are shown in Table 4-4.

SINGLE CHANNEL TRANSPONDER

The Single Channel Transponder is a dual-frequency band, frequency conversion, digital demodulation/remodulation communications transponder with integral command reception/execution capability. The primary function of this transponder is to provide a secure and reliable communications subsystem separate from the major communications payload. The transponder receives and processes messages and commands transmitted on either an SHF or UHF uplink. Messages are received on command, demodulated, modulated and retransmitted. Messages may also be stored in hardened transponder memory for repeated retransmission or the memory may be bypassed. Either of two redundent Transec's in the Digital Processor may be selected by TT&C to provide uplink instruction to the transmit and receive frequency synthesizer.

The critical nuclear survivability concerns with this type of subsystem are:

1. Transient Performance

a. "Fast" "Hard" Flip-Flop response

b. "Slow" "Hard" Flip-Flop response

c. Operating Parameters Data Register upset

d. Transec Command Interface upset

e. Data Processing recovery time

f. Shielded capacitors self-discharge response

Table 4-4. Power S/S transient performance requirements.

REQUIREMENTS	VERIFICATION
1. THE PRIMARY BUS VOLTAGE SHALL RECOVER WITHIN A SPECIFIED TIME.	IN-SITU MONITOR
2. SECONDARY BUS VOLTAGES BEING GENERATED BY THE DC/DC CONVERTERS SHALL RECOVER WITH 0.5 MILLISECONDS OF THE RECOVERY OF THE PRIMARY BUS VOLTAGE AND SHALL EXPERIENCE FOR NO MORE THAN 1.0 MILLISECOND AN OVERVOLTAGE CONDITION LIMITED TO A SPECIFIED MAXIMUM VALUE.	IN-SITU MONITOR
3. THE TIMING ERROR INDUCED IN THE BATTERY DISCHARGE TIMER SHALL BE WITHIN SPECIFIED LIMITS.	POST SHOT TELEMETRY INFORMATION
4. THE FAILURE DETECTOR SHALL NOT PROVIDE AN ERRONEOUS OUTPUT WHICH RESULTS IN A CHANGE IN MODE.	INFER FROM POST-SHOT TELEMETRY INFORMATION

g. "Hard" Logic Voltage response

2. Photocurrent Response, DC/DC Converter switching transistors' response

The specific survivability transient recovery and verification requirements are shown in Table 4-5.

COMMUNICATIONS SUBSYSTEM

The primary function of the Communication Subsystem (COMM S/S) is to provide a six-channel SHF communications repeater as the satellite payload. In providing this function, the communications subsystem performs all of the operations required to process the uplink signals and retransmit them to ground, including signal reception and jammer location, pre-amplification, channel routing and configuration, frequency translation, and power amplification for retransmission. The COMM S/S also provides a reference frequency to the Telemetry, Tracking and Command (TT&C) subsystem and the Single Channel Transponder (SCT).

Signal acquisition within the satellite coverage area is accomplished by the receive earth coverage antennas and a 61-beam Multibeam Antenna (MBA). Channel routing and configuration is established using electronically controlled RF ferrite switches. The transmitter outputs are multiplexed and fed to an earth coverage antenna, one of two 19-beam MBA's, or a gimbal dish transmit parabolic reflector antenna.

The Multibeam Antennas (MBA's) provide antenna coverage ranging from narrow beam to earth coverage. In the DSCS III satellite this capability is provided by one 61-beam receive antenna and two 19-beam waveguide lens transmit antennas. Associated with each MBA is the Beam Forming Network (BFN) under the control of the Attitude Control Electronics (ACE) which produces selected antenna patterns to fit the network of communicating users.

Table 4-5. SCT S/S transient performance requirements.

REQUIREMENTS	VERIFICATION
1. SYSTEM TIME MUST BE MAINTAINED WITHIN 50 MICROSECONDS.	IN-SITU MONITOR REQUIREING THE SIX-LINE INTERFACE BETWEEN THE SCT AND THE BEACON
2. THE POWER CONTROLLER's HARDENED BUSES SHALL RECOVER WITHIN 0.5 MICROSECONDS	INFER FROM INFORMATION OBTAINED IN VERIFYING REQUIREMENT NO. 1
3. OPERATING PARAMETERS SUCH AS: A. SPECIAL ACCESS CODES (WOD) B. UPLINK ADDRESS C. UPLING FREQUENCY MAY NOT CHANGE STATE UNLESS THAT SPECIFIC PARAMETER IS BEING UPDATED	POST SHOT TELEMETRY INFORMATION
4. OPERATING CONFIGURATION MAY NOT CHANGE UNLESS A VALID COMMAND IS BEING RECEIVED	POST SHOT TELEMETRY INFORMATION

Each MBA, therefore, consists of:

1. A waveguide lens array and
2. A BFN consisting of:
   a. A Central Electronics Box
   b. Waveguide ferrite phase shifters (receive MBA only)
   c. Waveguide ferrite variable power dividers
   d. Electronic drivers to set the states of the ferrites, and
   e. A phase transient controller.
3. A 61:1 RF switch (Receive MBA only)
4. An RF detector (Receive MBA only)
5. A digital controller (Receive MBA only)

The key nuclear survivability concerns with the MBA are:

1. Photovoltaic responses of the BFN driver transistors
2. Transient Recovery of the JLE Digital Controller and the RF Detector
3. "Hard" memory responses.

The specific survivability transient recovery and verification requirements are shown in Table 4-6. To verify the requirements on the BFN, the 19 BFN would be unpowered and the beam pattern monitored, while the 61 BFN would be powered and the satellite functional test determines burnout hardness.

Control of the receive MBA is accomplished by controlling the relative amplitude and phase of each of the MBA beams. Waveguide ferrite variable power dividers (VPD's) provide the relative amplitude control among the beams while waveguide ferrite phase shifters are used to control relative beam phase. In addition to the phase shifters, each waveguide lens of the receive MBA contains a passive coupler used to derive the RF signals for the Jammer Location Electronics (JLE). The BFN's are unpowered except during a reconfiguration

Table 4-6. MBA transient performance requirements.

REQUIREMENTS	VERIFICATION
BFN	
1. THE STATE OF THE BEAM PHASING FERRITES SHALL NOT CHANGE STATE WHEN UNPOWERED AND NO BURNOUT WHEN POWERED.	IN-SITU MONITOR

JLE TRANSIENT PERFORMANCE REQUIREMENTS

JLE	
1. THE DIGITAL CONTROLLER WHEN SET IN THE FAST SEARCH/SCAN MODE SHALL RE-INITIALIZE TO THE FAST SEARCH/SCAN MODE UPON THE RECEIPT OF A MASTER FRAME SYNC PULSE.	POST SHOT TELEMETRY INFORMATION
2. THE RF DETECTOR SHALL RECOVER UPON THE RECEIPT OF A MASTER FRAME SYNC PULSE.	INFER FROM DIGITAL CONTROLLER POST SHOT TELEMETRY

sequence. The Jammer Location Electronics (JLE) associated with the receive MBA is used to detect the presence and accurate location of any jammers. This information is transmitted to ground users via telemetry.

The key nuclear survivability concerns with the communications subsystem are:

1. Cavity SGEMP effects on RF Circuitry
   a. Low Noise Amplifier
   b. Tunnel Diode Amplifier Limiter
   c. Mixer
   d. FET Driver Amplifier
   e. RF Ferrite Switch
   f. Cavity Oscillators of the Frequency Generator phase locked loops
2. Transient Recovery
   a. Frequency Standard
   b. Frequency Generator phase locked loops
3. Radiation - Induced Mode Change
   a. RF Ferrite Switch
   b. Command/Telemetry IFU

The specific survivability transient recovery and verification requirements for the communications subsystem are shown in Table 4-7.

TELEMETRY TRACKING AND COMMAND SUBSYSTEM

The Telemetry Tracking and Command (TT&C) Subsystem provides formatting, encryption, and transmission of satellite telemetry; reception; filtering; and transmission of the ranging signal; and reception, decryption and decoding of commands. The subsystem is comprised of independent redundant S-Band and redundant X-Band (SHF) links. The S-Band links use cross-dipole antennas mounted on opposite sides of the satellite to provide near spherical coverage.

Table 4-7. Comm S/S transient performance requirements.

REQUIREMENTS	VERIFICATION
1. COMPONENT RECOVERY TIME OR TIMING ERROR ALLOWED	
A. LNA - 85 MILLISECONDS	IN-SITU MONITOR
B. TDAL - 85 MILLISECONDS	IN-SITU MONITOR
C. FREQUENCY STANDARD - ERROR - 10 MICROSECONDS	IN-SITU MONITOR
D. FREQUENCY GENERATOR - 85 MILLISECONDS	IN-SITU MONITOR
E. FET DRIVER AMPLIFIER - 85 MILLISECONDS	IN-SITU MONITOR
F. TLS - 85 MILLISECONDS	IN-SITU MONITOR
G. TWTA - 85 MILLISECONDS	IN-SITU MONITOR
2. RF SWITCHES SHALL NOT CHANGE STATE	INFERRED FROM IN-SITU MONITOR
3. SIGNAL DIRECTIVITY AND PORT-TO-PORT ISOLATION SHALL REMAIN WITHIN SPECIFICATION	INFERRED FROM IN-SITU MONITOR

The S-Band links are used primarily for normal command and control of the satellite support subsystems. The secure SHF links are used primarily for command and control of Communications Payload. Both the S-Band and SHF links, however, can provide total satellite telemetry and command capability. SHF Command modulation is received through Communications Subsystem Receive MBA or Earth Coverage Receive Antennas.

All TT&C components, except forward and aft antennas, are located on the satellite North and South Panels. The important survivability concern with this subsystem is that the spacecraft command processing might upset and cause erroneous commands to be executed. The specific survivability transient recovery and verification requirements for this subsystem are shown in Table 4-8.

ATTITUDE CONTROL SUBSYSTEM

The Attitude Control Subsystem (ACS) provides the functions of attitude sensing, telemetry and command data processing and storage, signal processing, mode switching and drive signals for actuation, solar array control and gimballed antenna drive. These functions are provided for the following modes:

1. Initial Drift Orbit and Acquisition
2. Operational Mode
3. Station Change Mode
4. Re-acquisition Mode
5. Failed Battery Mode

The ACS consists of the Attitude Control Electronics (ACE), an Earth Sensor, two Sun Sensor Assemblies, a Rate Gyro and four Reaction Wheels. The ACE provides all the data processing, the control computations and the electronics support for the ACS. The functional elements of the subsystem are as follows:

Table 4-8. TT&C S/S transient performance requirements.

	REQUIREMENTS	VERIFICATION
1.	POWER "UP" FROM THE STANDBY MODE TO THE VERIFY AND/OR EXECUTE MODES SHALL NOT OCCUR.	IN-SITU MONITOR
2.	THE POWER SWITCHING CONTROL LOGIC SHALL NOT CAUSE A CHANGE FROM "OFF" TO "ON" OR VICE-VERSA.	POST SHOT TELEMETRY INFORMATION
3.	THE S-BAND TRANSPONDER SHALL RECOVER AUTOMOUSLY WITHIN THE SPECIFIED ACQUISITION TIME OF 8 TO 10 SECONDS.	IN-SITU MONITOR
4.	THE SHF/S DOWNCONVERTER SHALL RECOVER WITHIN THE NORMAL PHASE-LOCKED-LOOP ACQUISITION TIME.	IN-SITU MONITOR
5.	THE AUTOMATIC GAIN CONTROL STEP ATTENUATOR CONTROLLER SHALL CAUSE A CHANGE OF NO MORE THAN ONE STEP.	POST SHOT TELEMETRY INFORMATION
6.	THE S/IF DOWNCONVERTER'S HARDENED BUS SHALL RECOVER WITHIN 0.5 MICROSECONDS.	INFER FROM POST SHOT TELEMETRY INFORMATION
7.	THE S/IF DOWNCONVERTER SHALL RECOVER WITHIN THE NORMAL PHASE-LOCKED-LOOP ACQUISITION TIME.	IN-SITU MONITOR
8.	THE BEACON SHALL RECOVER WITHIN THE NORMAL PHASE-LOCKED-LOOP ACQUISITION TIME.	IN-SITU MONITOR
9.	COMMAND PROCESSING SHALL RECOVER WITHIN THE ASSOCIATED PHASE-LOCKED-LOOP RECOVERY TIMES.	PROCESS A COMMAND WITHIN 8 TO 10 SECONDS
10.	TELEMETRY PROCESSING SHALL NOT CHANGE OPERATING MODE AND SHALL RECOVER WITHIN THE ASSOCIATED DATA SAMPLING RATE.	POST SHOT TELEMETRY INFORMATION

1. Sensing - Senses the satellite attitude with respect to the earth and the sun. Senses yaw body rates, reaction wheel speeds, solar array position and a nuclear event

2. Sensor Signal Processing - Provides buffering, multiplexing, A/D conversion and digital processing of sensor data

3. Attitude Correction - Samples and processes sensor data and performs error calculations in accordance with control laws

4. Wheel Control - Provides actuation signals to the reaction wheels and thrusters

5. Command Processing - Receives and executes ground commands

6. Telemetry Processing - Provides telemetry data for transmission to the ground

The attitude control function is implemented with two digital loops, which update the state vector of the vehicle. The updates are performed every sixteen seconds during the operation mode and two seconds on all the other modes. The specific survivability transient recovery and verification requirements for this subsystem are shown in Table 4-9.

ADDITIONAL MEASUREMENTS

In addition to the measurements whose purpose is to verify specific spacecraft survivability response, there are several additional classes of measurements that are required. These additional classes of measurements include:

electronics diagnostics,

background,

solar array (S/A) currents,

S/A shaft currents,

external fields/shield currents,

antenna responses,

port of entries,

internal fields/shield currents, and

component box core currents

cable shield currents.

Table 4-9. Attitude control S/S transient performance requirements.

REQUIREMENTS	VERIFICATION
1. THE EARTH SENSOR'S PITCH, ROLL AND EARTH PRESENCE OUTPUTS SHALL PROVIDE VALUE DATA WITHIN 5 SECONDS	IN-SITU MONITOR
2. THE SUN SENSOR'S PITCH AND ROLL OUTPUTS SHALL PROVIDE VALID DATA WITHIN 30 SECONDS	IN-SITU MONITOR
3. THE REACTION WHEEL TACHOMETER OUTPUT SHALL RECOVER TO NORMAL OPERATION WITHIN 15 MILLISECONDS	IN-SITU MONITOR
4. THE REACTION WHEEL CONTROL FUNCTION SHALL CAUSE AN ADDED S/C POINTING ERROR OF NO MORE THAN 0.07 DEGREES	IN-SITU MONITOR PLUS ANALYSIS/SIMULATION
5. THE EVENT DETECTOR OUTPUT SIGNAL SHALL: A. DISABLE WRITING INTO THE HARDENED MEMORY WITHIN 25 MICROSECONDS B. INITIATE OPERATION OF THE "HARD" +5 VOLT SWITCH C. INITIATE THE CPU EVENT RECOVERY ROUTINE D. PROVIDE AN INPUT TO THE THRUSTER DRIVE AND BFN INTERFACE FUNCTIONS	INFER FROM POST SHOT TELEMETRY INFORMATION
6. THE THRUSTER DRIVE FUNCTION SHALL DISABLE THRUSTER FIRING WITHIN 100 MICROSECONDS	IN-SITU MONITOR
7. THE BFN INTERFACE FUNCTION SHALL: A. PREVENT APPLICATION OF POWER TO AN "OFF" BFN B. REMOVE POWER FROM AN "ON" BFN WITHIN 100 MILLISECONDS	IN-SITU MONITOR
8. THE SOLAR ARRAY DRIVE FUNCTION SHALL CAUSE A SOLAR ARRAY DRIVE SHAFT ROTATION OF NO MORE THAN 1 DEGREE	IN-SITU MONITOR
9. THE GDA DRIVE FUNCTION SHALL CAUSE A GDA POSITION ERROR FOR MORE THAN 15 MILLISECONDS	IN-SITU MONITOR
10. THE COMMAND AND TELEMETRY PERIPHERALS SHALL PROVIDE NORMAL OPERATION WITHIN 15 MILLISECONDS	VERIFY PROCESSING A COMMAND WITHIN 8 TO 10 SECONDS
11. THE ATTITUDE CONTROL ELECTRONICS HARDENED BUS SHALL RECOVER WITHIN 0.5 MICROSECONDS	INFER FROM POST SHOT TELEMETRY INFORMATION

Electronic diagnostics measurements are used to augment component box recovery measurements. This class of measurements would include, power supply voltages, surge currents and other component box functional measurements. Background measurements are used to validate test data and establish background noise levels. There should be a background measurement for each unique type of instrumentation/sensor on the spacecraft. Solar array currents are measured to characterize the surges that are injected into the spacecraft because of the response of this component. The rest of the data classes are associated with SGEMP effects. This data is needed to extrapolate the satellite response to higher level threats.

For satellite systems the SGEMP threat can be classified into four response categories: External SGEMP, Cavity IEMP, Cable Direct Drive, and Black Box SGEMP. These categories are defined in the following paragraphs.

External SGEMP - Electron emission from the irradiated external surfaces of a satellite produces a space current which drives electric and magnetic fields in the neighborhood of the satellite and currents within the structural surfaces of the satellite. These fields and currents may couple to electrical systems directly, or indirectly through penetration into the satellite interior.

These external fields and structure currents would be measured to verify S/C response models and to characterize the environment to which the S/C is exposed.

Cavity IEMP - Photons which penetrate into the satellite interior drive electrons from irradiated interior electrical cabling. The magnitude of the coupling from these cavity fields should be verified by test. The cavity IEMP would be characterized by field measurements in each of the

major satellite cavities. Shield Currents would also be measured in each of these cavities to determine the amo :t of coupling between the cavity fields and the satellite electrical harnesses. This data would be used for scaling purposes. A determination of the shielding effectiveness of the satellite cables would probably not be practical in this test without heavily instrumenting because of their multiended characteristic and variation in load impedance.

Cable Direct Injection - Photons which penetrate into electrically shielded cables cause electron emission from the interior surfaces of cable shields and exterior surfaces of cable conductors. This charge displacement causes replacement currents on the cable conductors which propagate to interfacing circuitry. The current level on cable cores due to photon direct injection must be verified by test. Because the cables are also multi-ended it is difficult to determine the basic cable direct injection source term in this configuration. Thus, these effects would be best measured using a special cable installed in the S/C for this purpose or these effects would be verified in a separate cable test.

Black Box SGEMP - Photons which penetrate into electrical component boxes causes bulk currents within dielectrics and electron emission from the internal surfaces of boxes and the external surfaces of boxes and the external surfaces of conductors. The resulting currents cause replacement currents and fields which couple to wiring and printed circuitry. This effect is particularly significant in component boxes such as RF amplifiers which utilize high atomic number plating materials in their internal construction. The verification that these effects do not cause any upset are inferred from spacecraft telemetry and the internal box measurements. Measurements of the internal box fields and currents could probably not be done without either

perturbing the SGEMP response or perturbing the box operation since these are RF boxes. Thus scaling data could not be obtained and a burnout test would have to be performed at the box level.

The net effect of the various SGEMP responses is to establish an electrical pin threat (pin specification) to the interface circuits of component boxes as developed in Figure 4-5, which shows the contribution of the various SGEMP response mechanisms. The diamond-shaped boxes in this figure depict design requirements of the satellite system that should be verified by test. The key hardening features required to meet these design requirements may be different on different systems, however, the hardness to each of the SGEMP response mechanisms (i.e., External SGEMP, Cavity IEMP, Cable Direct Injection and Box IEMP) should be verified.

The data from the SXTF tests could be scaled to threat level in order to verify the component box pin specification. The pin specification determines how hard the system should be in terms of expected current and voltage waveforms. The pin specification is also used to set the test level for the component box current injection test. The SXTF tests augmented by cable photon tests and cable shielding effectiveness tests verify that the pin specification is correct. These tests will be used to verify the various analyses and models used to determine the pin specification.

Core currents would probably be measured rather than pin currents for a number of reasons. Satellite cables typically are multi-wire cables containing up to one hundred wires. The cable direct injection on a wire is a function of its position within the cable. The position of each wire is uncontrolled and varies along the length of the cable, thus the SGEMP response of a particular wire would vary from spacecraft to spacecraft.

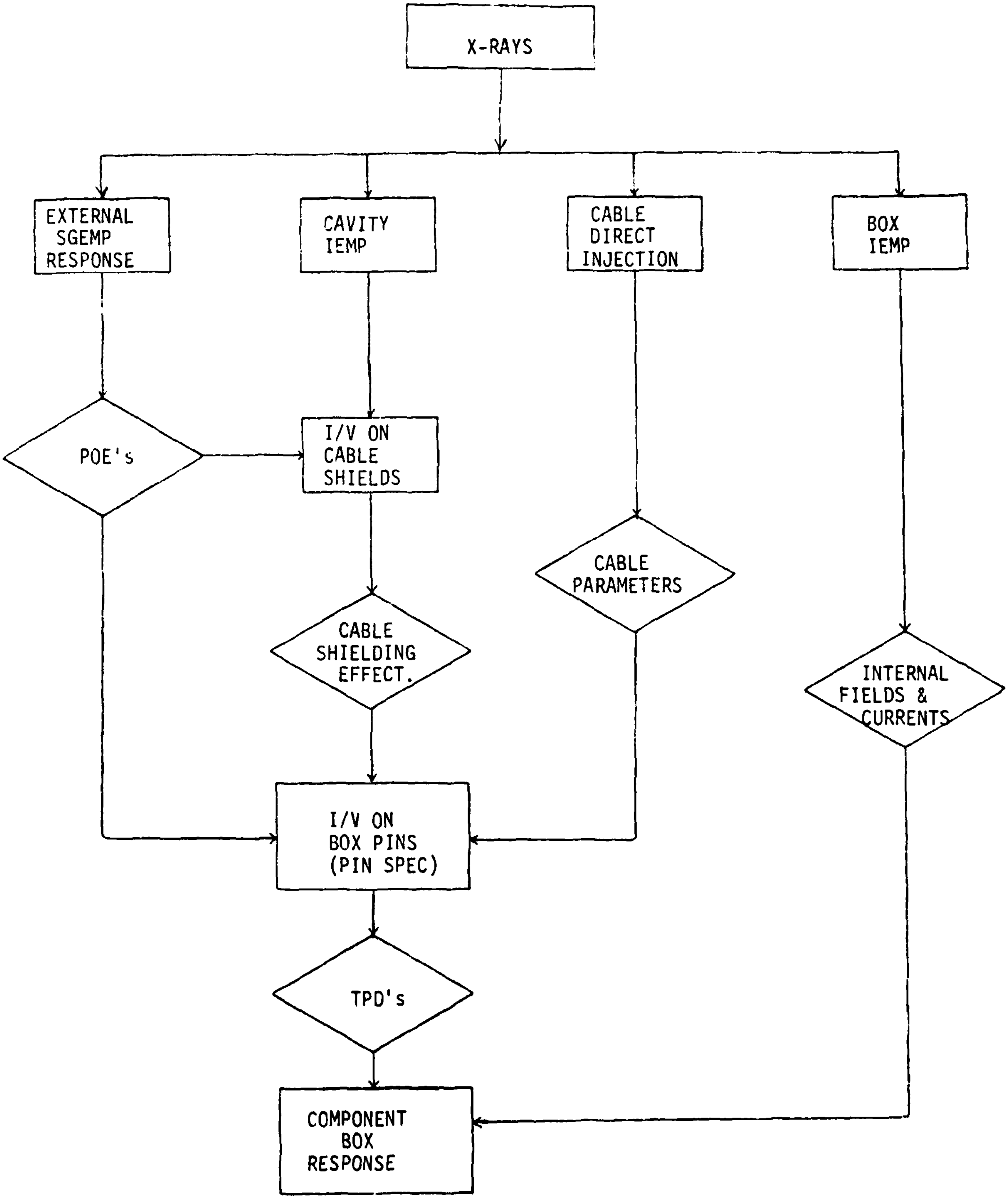


Figure 4-5. SGEMP response mechanisms effects on component boxes.

Therefore, core currents will be measured and scaled except for a few places where there are special concerns (i.e., command lines). Pin current measurements would, however, be made at each transmitter output to characterize the Antenna responses.

4-4 INSTRUMENTATION REQUIREMENTS.

A summary of the number of instrumentation channels required for a test of a satellite by measurement function is shown in Table 4-10. This assumes that the SXTF exposure level is less than the threat level and scaling of the SGEMP data is required. The required frequency response for the instrumentation is shown in Table 4-11. Most of the instrumentation channels fall into two classes of frequency response. Many of the S/C recovery measurements require a frequency response from DC to 10KHz while the SGEMP measurements require a frequency response up to 500 MHz. Data for all of these channels would be transmitted over fiber optic links in order to provide the needed satellite electrical isolation. The FOL transmitters would probably be located in a "splice box" located on the bottom of the S/C. Multiplexers could be used for the low frequency channels to reduce the quantity of fiber optic links required. It is anticipated that installation of the sensors would be similar to the installation techniques used on STARSAT for the Huron King test.

Table 4-10. Baseline SXTF instrumentation requirements.

MEASUREMENT FUNCTION	INSTRUMENTATION POINT
1. S/C SURVIVABILITY REQUIREMENTS	
- POWER S/S	4
- PAYLOAD S/S	11
- MBA S/S	19
- SCT S/S	7
- ACS S/S	47
- TT&C S/S	9
	97
2. ADDITIONAL DATA	
- ELECTRONICS DIAGNOSTICS	20
- BACKGROUND	4
- SOLAR ARRAY I	2
- SHAFT I	2
- EXT. FLD./SHLD. I	6
- ANTENNA RESPONSE	6
- P.O.E.'S	10
- INT. FLD./SHLD. I	10
- BOX CORE I	20
	80

Table 4-11. Sensor frequency response and configuration requirements for SXTF test.

S/C MEASUREMENT	QUANTITY
EXTERNAL TO S/C	
f DC-10KHZ	19
f $<$ 10MHz	1
f $<$ 500MHz	24
INTERNAL TO S/C	
f DC-10KHZ	70
100HZ $<$ f $<$ 100KHZ	10
f $<$ 10MHz	11
f $<$ 500MHz	42

NOTES: 1. FREQUENCY RESPONSE IS FOR DATA LINK RECORDER.

RF → LF DETECTORS NOT SPECIFIED HERE

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SXTF DESIGN SUPPORT STUDY.(U)
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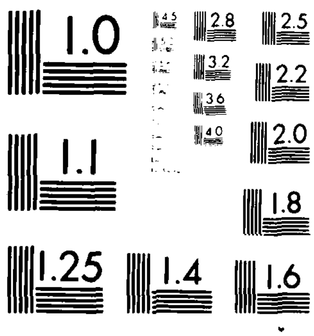

MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS 1963 A

SECTION 5

SATELLITE SYSTEM TESTING

5-1 SATELLITE TESTING.

Any satellite system test program such as envisioned for SXTF should place major emphasis on establishing a comprehensive performance evaluation program. The system level performance, subsystem compatibility and verification of specific mission requirements are determined from the particular satellite technical specification derived from the satellite procurement specifications. Performance testing is divided into segments: Full-up Electrical Systems Test and Confidence Testing. This permits a cost-effective approach to satellite testing without compromising the integrity of the spacecraft.

It is recommended that, prior to the Satellite X-Ray Test, a full-up electrical systems test be performed at the factory and confidence type tests be utilized while at the SXTF. This will provide sufficient information on the health of the satellite after each exposure while minimizing the total satellite test time at the SXTF.

A full-up electrical systems test is performed after all subsystems have been carefully integrated into the satellite. It provides a complete verification of performance, establishes the calibration of all instruments and sensors, illustrates compatibility between subsystems and constitutes the performance baseline for subsequent testing. This test is extremely important. Because of the extensive nature it is a very time consuming test. For complex satellites such as the DSCS III, 45 shifts are required to complete this type of test. To become cost-effective this test is conducted after integration of all subsystems and at the completion of all environmental testing. A reduced version of this test is the Electrical System Baseline.

An Electrical System Baseline test is performed to evaluate satellite performance during and between major environments; i.e. during thermal vacuum testing, intra axis vibro-acoustics and before and after environmental exposures. Fifteen shifts are required to complete this task. This test is directly traceable to the full-up electrical system test and would be performed before and after SXTF testing.

A much reduced Electrical System Baseline Test is a Confidence Test which guarantees the health of the satellite by functionally checking the various subsystems. This is especially useful when a verification is required after the satellite has been transported, or during field site testing. The confidence test as applied to the DSCS III satellite takes approximately 3 shifts.

It is anticipated tha a confidence test would be performed a number of times during the SXTF exposure sequence. The exact number of these tests would be determined on experience gained during SXTF testing, but as a minimum it should be done between changes in the satellite orientations.[2]

In general, the satellite will be commanded to a known mode of interest prior to a "shot". Depending upon the satellite and the operating sub-systems this turn on may take up to 4 hours. The satellite telemetry will be monitored for approximately ten minutes in a "steady state" mode before and after the "shot". Though the use of real time satellite "Status Program" computer monitoring, it is possible to immediately determine if any sub-system has changed state as a result of the radiation test. This provides a "static" test of the systems survivability compared to the dynamic system exercising used in the confidence and electrical system test. A typical satellite test flow illustrating the test sequences from factory to SXTF and return is shown in Figure 5-1. Figure 5-2 and 5-3 show typical areas required to support this type of satellite testing program at SXTF.

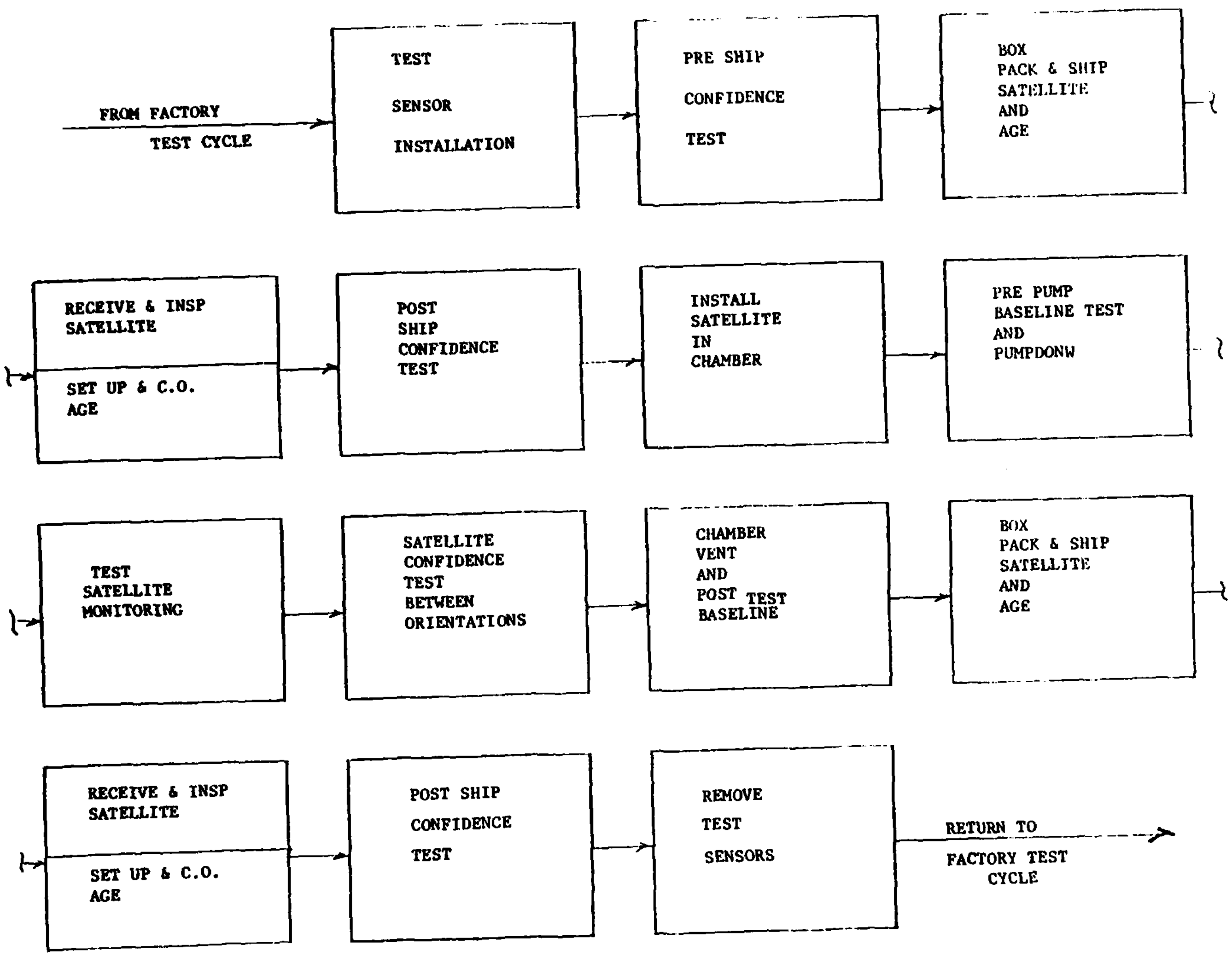

Figure 5-1. SXTF test cycle.

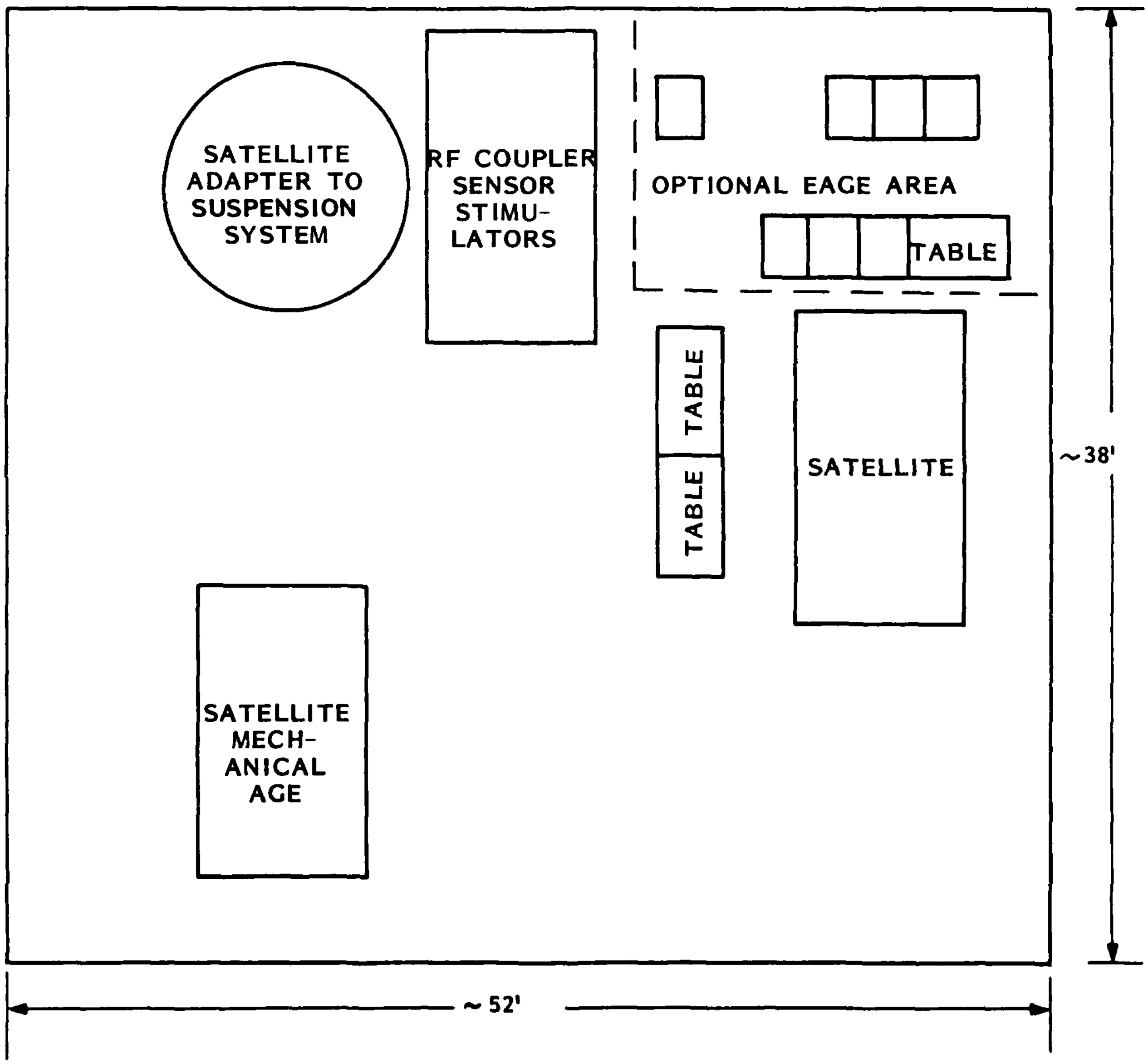

Figure 5-2. Preparation area typical footprint.

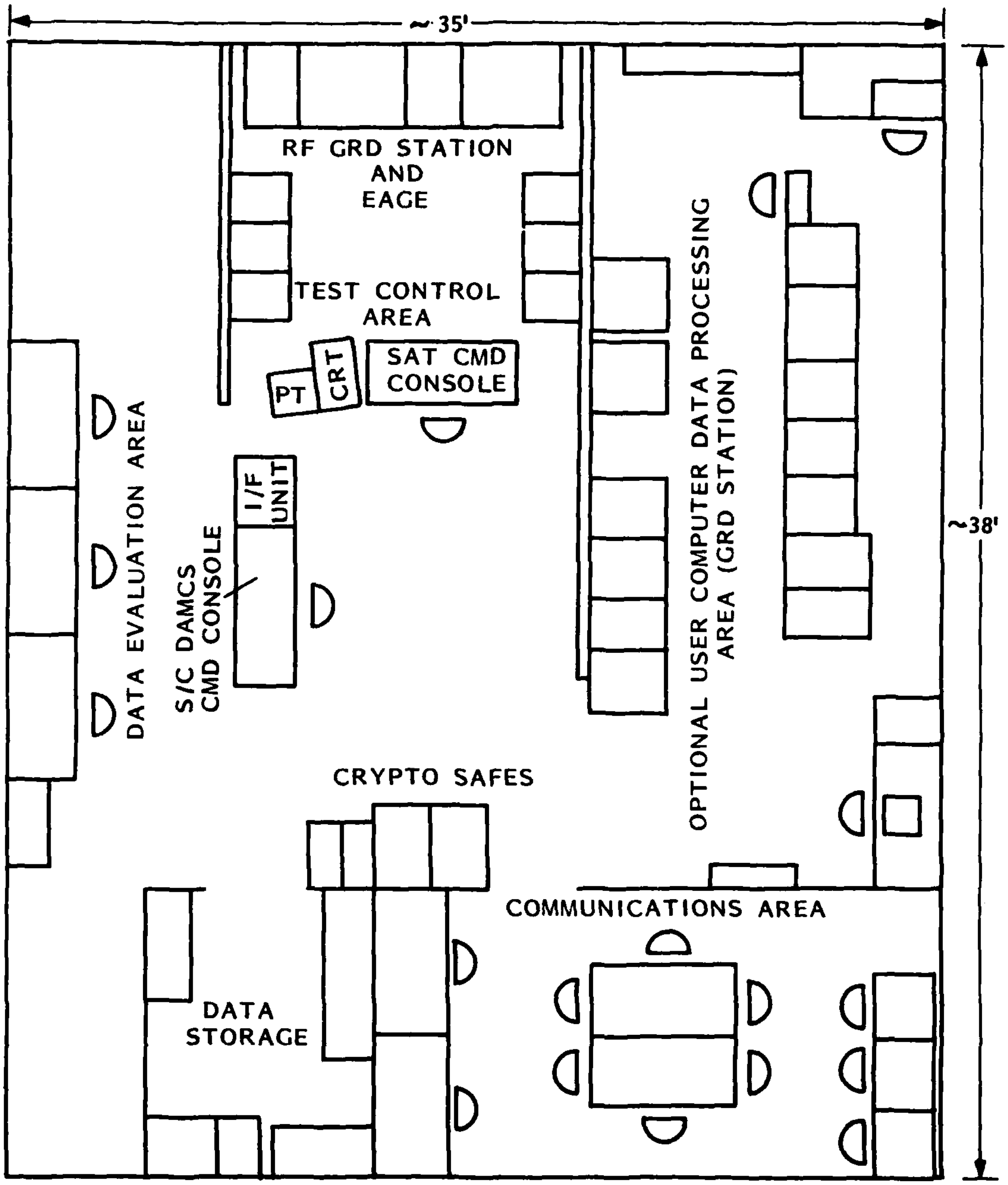

Figure 5-3. Ground station area footprint.

5-2 SATELLITE DATA EVALUATION.

A major concern with any off site test activity is how the satellite data is acquired, processed and evaluated. Data acquisition on the DSCS III satellite requires both an X-Band and an S-Band ground station. Processing of the data is performed by a computer network and placed into a format for an effective evaluation of satellite parameters. A "stand alone" and a "dependent" configuration approach to the data handling and evaluation problem are discussed for use at SXTF.

The "Stand Alone" configuration implies that all the equipment which is necessary to test the satellite and process all data on an off-site facility will be required at SXTF. This entails a major shipment of all ground support equipment, installation and complete checkout prior to application of power to the satellite. Shipments of this magnitude create additional problems, especially in the area of logistics. Staffing requirements are also increased since computer operators, software personnel and data analysts are needed at the facility. This approach would be costly and would require an approximate one month additional tie up of the facility.

The "Dependent" configuration differs from the "Stand Alone" in that all the data processing and evaluation is performed at the contractor's facility. This requires a Modem link installed at the SXTF and the same system installed at the contractor's facility. The data from the satellite is acquired by the ground station and is then transmitted over the telephone lines, via the Modem, to the contractor's facility. This approach is more cost effective in that less equipment, set-up time, personnel and facility occupancy is required. This approach was demonstrated with the Starsat vehicle while undergoing tests at the Nevada Test Site. It is recommended that this type of data transmission system be instituted at the test facility.

5-3 TRANSPORTATION AND HANDLING.

The overall transportation effort is the responsibility of the manager of Traffic Services for the satellite contractor. Support in implementing and coordinating this effort, however, will be required from the satellite Program Office and the DNA SXTF test coordinator or conductor.

Prior to "packing" the satellite will undergo a preshipment confidence test and inspection. The completely assembled satellite will then be sealed in a protective "humidity" bag and secured in an environmental cover.

All AGE will be secured and protected in accordance with manufacturer's instructions.

The Satellite and its AGE should be transported overland in/on commercial vans/trailers equipped with an "air ride" suspension system, by a responsible carrier who has a proven record of success in prior movements of aerospace prime hardware.

For transportation purposes, classified materials should be treated separately and distinct from non-classified materials. Where practical, all classified items will be disassembled from major components and shipped separately. Shipments of all classified materials and the satellite, if so designated will be coordinated by the Program Custodian, the Air Force Cryptologic Depot and the Armed Forces Courier Service.

Contractor personnel would be required to escort the shipment in conjunction with the Armed Forces Courier Service, if a courier is required. The satellite contractor would also provide the "Wagon Master" who will have overall authority while en route. On site security would be the joint responsibility of the satellite contractor and the SXTF test conductor, the implementation of which is not yet determined.

Upon arrival at the SXTF, the satellite and AGE would be unpacked and prepared for test. In support of this activity, the SXTF would be required to provide a high bay area with crane for unloading the satellite and suitable fork trucks, dollies, etc. for unloading and positioning the ground support equipment (GSE). Additionally, the SXTF would provide warehousing or storage facilities for the environmental cover, packing crates, pallets, etc, until required for return shipment.

After unpacking the satellite would be moved into the "prep area" and a receiving inspection performed. When the GSE is ready, a receiving confidence test will be performed. This test would satisfy that the satellite sustained no damage during transport, and assure that all GSE and site to factory data links are operational. Upon completion of this test, the satellite can be installed in the test chamber.

After unpacking and during the satellite inspection, the AGE would be installed, interconnected and checked out in preparation for satellite powered testing. SXTF to factory and intra-SXTF data links would also be verified.

Handling of all satellite associated equipment should be performed by contractor personnel under the direct supervision of the contractor's handling supervisor.

Handling of the satellite by the contractor would be performed by a trained System Test handling crew. In those areas where the satellite interfaces with facility operated equipment, e.g. primarily cranes, the facility operator would control the satellite to within some small distance, e.g. six inches, of any mating surface and the satellite system test team would complete the move using "hydra sets" or other suitable micropositioners. The suspension of the satellite in the chamber would be performed by the satellite test team, assisted by facility personnel where necessary to interface to facility hardware.

5-4 SXTF TEST CYCLE WINDOW.

There are two windows in the satellites total test cycle that are recommended for SXTF use, depending on the model of satellite to be tested. These windows are shown in the satellite intergration and test flow diagram, Figure 5-4. The dashed blocks show the added elements for either of the two models of satellite. The satellite environmental testing involves thermal balance, thermal vacuum, vibration and acoustic testing while the mechanical tests involve weight, center of gravity and mass properties evaluations.

A qualification, protoflight or research model satellite would be tested late in the test cycle after integration, environmental and mechanical acceptance tests have been completed. This would ensure that the satellite design is proven and acceptable prior to SXTF testing. Satellites of these model types would either not fly or would require complete factory refurbishment prior to designation as flight hardware. Satellites of these types could be fully instrumented for an SXTF test.

Flight Model satellites on the other hand would be available for SXTF evaluation immediately after subsystem integration. A minimal internal instrumentation perturbation to the vehicle would probably be allowed. This approach would allow the full battery of acceptance tests to be performed after SXTF exposure to determine any detrimental effects.

Production model satellites would be handled in the same timeframe and much the same as the Flight Models, except that no internal instrumentation would be allowed. The test would be performed on a pass/fail basis and only on randomly selected units.

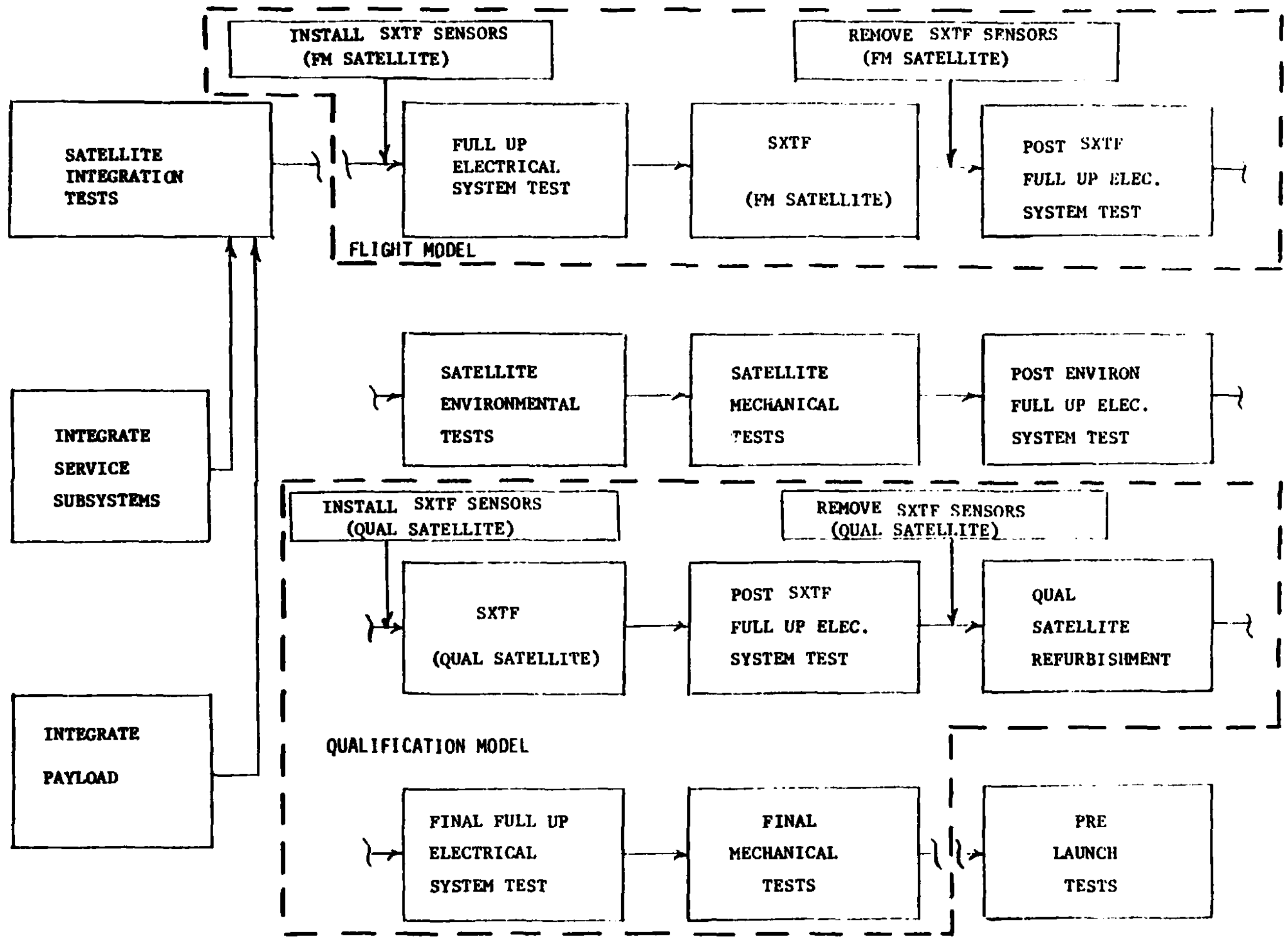

Figure 5-4. Vehicle test cycle.

SECTION 6
CHAMBER DESIGN

6-1 MATERIAL CHARACTERISTICS.

All materials have characteristic vapor pressures which vary with temperature. In order to achieve low pressures in the range of 10^{-5} torr for the SXTF application, no material exposed to vacuum must have a vapor pressure such that it will volatilize at the anticipated operating temperature to which it will be subjected at these low pressures. It is perhaps obvious that materials which are recognized as having a low boiling point, such as water, grease and ordinary oils could be expected to give trouble in a high vacuum system. What is not so obvious, however, is that materials which one ordinarily does not think of as being readily volatilized will give off vapors which will impact the integrity of the vacuum system. Materials of construction for vacuum systems must be carefully selected in order to prevent volatilization of various gaseous components of the material leading both to breakdown of the materials and to an inability to achieve hard vacuum.

For large chambers employing large cryogenically cooled liquid nitrogen panels, the ability to condense most of these vapors normally will enhance achievement of 10^{-5} torr. A greater concern is the vapors which may condense on a test spacecraft and result in detrimental effects on spacecraft operation. From an SXTF view point, therefore, the concerns should center on facility cleanliness, its measurement and maintenance.

Contaminants may be introduced into a space-environment-simulation chamber during the construction phases, test preparation phases and during the phases of satellite testing. Once introduced, these contaminants can cause detrimental effects on some satellite subsystems.

In relation to satellite testing, contaminants are categorized as either particulate and molecular. Particulate contamination consists of dust, lint, rust, human epidermal cells, metal particles, etc. This form of contamination normally is experienced when the chamber pressure is high enough to transport through convective air circulation. This is during satellite test preparation, initial pumpdown and final repressurization. The presence of particulate contamination is generally not experienced under high vacuum conditions unless there is an internal source.

Molecular contamination is a more serious problem during high vacuum conditions than particulate contamination. Particulates can normally be removed and seldom cause irreversable detrimental effects to the satellite. Molecular contamination consists of chemical compounds that can be deposited, throughout the chamber test volume. When these substances condense on optical surfaces, solar cells, passive thermal control coatings and the like, they may interfere with the normal operation of these elements. Again, the degradation caused by the contamination is normally not permanent. Experience has shown, however, that full restoration may not be achievable if contaminated optical surfaces are irradiated with ultra-violet or other particles such as electrons or protons.

Materials that produce molecular contamination in a space-simulation chamber are commonly from outgassing products of paints, epoxies and insulations. They can also be caused by improper operation and poorly designed oil-diffusion pumping systems.

Satellites have often been a source of contamination and as a result, controls have been established through the use of preferred materials lists which are applicable to satellite hardware[3].

From an SXTF viewpoint, a contamination control plan is required to minimize contamination during construction. It would be responsible for contamination control through the use of proper materials and construction techniques, developing proper SXTF operating procedures, selecting SXTF cleaning methods and determining insitu contamination monitoring and measurement techniques.

6-2 PUMPING SYSTEM.

The major identifiable potential sources of molecular contamination in the AEDC system is the diffusion pumps. Use of proper system design and operation has reduced back streaming concerns to acceptable levels. Recent experience, however, indicates a risk is still prevalent by the mere presence of diffusion pump oil. Complete removal of the SXTF diffusion pumps is recommended. Development of cryogenic pumps is currently at a point where cryogenic pump modules can be readily adapted to chamber interfaces which presently house diffusion pumps.

The AEDC Chamber is equipped with 20°K helium shrouds used in conjunction with its LN_2 cryopanels. Temperatures are obtained with two large helium refrigerators, each having a 4KW capacity. Operation of these panels is expensive although they provide high pumping speeds for air, they may add significantly to the gas load through helium leakage. Helium and hydrogen cannot be "pumped" through condensation on 20°K surfaces.

Replacement of the diffusion pumps with their total 2 X 10^5 liter/sec air pumping speed would require nominally eight modules. Current module prices are in the order of $50K each. Net pumping of air in the order of 10^5 liter/sec allows a total chamber leak rate of 1 torr-liter/sec when operating at 10^{-5}torr. This is equivalent to approximately 3.6 cc/sec helium. Normal leak rates for large chambers can be limited to much less than this - on the order of 10^{-3}cc/sec. The installation of cryopumps will

assure that molecular contamination will not occur from diffusion pump oil.

Since cryogenic pumps are not effective until the chamber pressure is down to approximately 10^{-4} torr, additional oil free pumping may be required. This would provide a transition from mechanical blowers to cryopumps and is highly dependent on chamber leakage. Turbo-molecular pumps can provide this capability if required.

Chamber leakage is of primary importance in determining not only cryopumping requirements but all roughing and transitional pumping requirements. The AEDC chamber leakage should be characterized and considered with respect to an acceptable net limit after modification. When this required maximum leak rate is established, measurement procedures can be established to assure conformance to specifications.

6-3 CLEANLINESS REQUIREMENTS.

Typical satellite cleanliness requirements are based on specific sensitive areas. These are delineated by both MIL STD-1246A and NASA-SN-C-0005. Minimum surface cleanliness levels acceptable for those components identified critical to spacecraft performance for DSCS-III, for example, are specified below:

COMPONENT	SURFACE CLEANLINESS LEVEL (See Table 6-1)
OSR's	1000 A
*Thermal Blankets (MLI)	1000 B
Solar Arrays	1000B
Optics/Sensors	1000B
Structure	1000D
Vents	Eliminate or close where critical surfaces listed above are in field of view.

Table 6-1. Surface cleanliness levels. (From NASA SN-C-0005)

Level	Particle Size, Range, Micrometers (Note 1, Note 2)	
25	5	Unlimited (Note 3)
	5 thru 15	19
	15 thru 25	4
	25	0
50	15	Unlimited (Note 3
	15 thru 25	17
	25 thru 50	[illegible]
	50	0
100	25	Unlimited (Note 3)
	25 thru 50	oS
	50 thru 100	11
	100	0
150	50	Unlimited (Note 3)
	50 thru 100	47
	100 thru 150	5
	150	0
200	50	Unlimited (Note 3)
	50 thru 100	154
	100 thru 200	16
	200	0
250	100	Unlimited (Note 3)
	100 thru 200	39
	200 thru 250	3
	250	0
300	100	Unlimited (Note 3)
	100 thru 250	93
	250 thru 300	3
	300	0
500	100	Unlimited (Note 3)
	100 thru 250	1073
	250 thru 500	27
	500	0
750	250	Unlimited (Note 3)
	250 thru 500	205
	500 thru 750	9
	750	0
1000	500	Unlimited (Note 3)
	500 thru 750	34
	750 thru 1000	5
	1000	0

Level	Maximum Quantity NVR* per 0.1 Square Meters (1 sq. ft.) (Note 1)
A	1 mg
B	2 mg
C	3 mg
D	4 mg

***Non Volatile Residule**

EXAMPLE - Level 300 B would be particulate level 300 plus NVR level B.

Table 6-1 (continued)

NOTES

(NOTE 1) Particulate and Non-volatile residue (NVR) allowables are based on 0.1 square meters (1 square foot) of surface area. Flush fluid quantity for sampling shall be 100 milliliters (ml.) per 0.1 square meters (1 square foot) of surface area. Small parts should be grouped together to obtain 0.1 square meters (1 square foot) of surface area.

(NOTE 2) Maximum quantity per 1.0 standard cubic meters (35 standard cubic feet) of effluent gas when systems are being evaluated by purging. If feasible, the sampling must be accomplished at maximum system operational flow rate.

(NOTE 3) Unlimited means particulate in this size range is not counted; however, if the accumulation of this silt is sufficient to interface with the analysis, the sample shall be rejected.

SECTION 7

SATELLITE POWERING

7-1 INTRODUCTION.

In order to adequately define the requirement or specification for a system which will be designed to supply power to a satellite on a non-interference basis with the satellite response to x-ray or electrostatic charging tests, it is necessary to characterize and understand the satellite's electrical power and thermal dissipation systems.

There are two satellite subsystems which represent sources of power normally available to a satellite electrical power distribution system (EPDS): the solar cell-array and the batteries. Certain vehicles may use a nuclear powered supply in place of the solar cell array because of mission objectives or the particular orbit. On the geosynchronous DSCS III vehicle there are four electrical solar cell array panels each about 29 square feet in area and able to supply a maximum of about 309.5 watts at 28 volts DC to the Power Regulation Unit (PRU) at beginning of life (BOL) with normal full solar illumination. The majority of the solar cell circuits are configured in a matrix of 81 series by 5 parallel-connected 10 ohm-cm cells. The positive connection of each matrix is individually wired to a Shunt Dissipator (SD) located on the solar array yoke. At the Shunt Dissipator the solar array (SA) circuits are diode junctioned into a common +28v bus arrangement which is transferred through redundant sliprings to the PRU. Table 1 represents the expected satellite load power demand under on-orbit conditions. The table represents

average power requirements; instantaneous peaks can exceed the average up to a peak of 1500 watts maximum.

During eclipse peak power is limited to 1500 watts but battery capacity is available for higher power for a length of time determined by the battery depth of discharge. During summer solstice (lowest SA power in the year) the load on the PRU is about 824 watts with an additional 45 watts from the SA going to recharge the batteries. During equinox season the average PRU power load is 854 watts with an additional 118 watts going to battery recharging from an almost 70% (end-of-life) depth of discharge during the longest eclipse with maximum predicted satellite load (SCT active).

Table 7-1. Load power demand (watts).

	ON STATION					
			EQUINOX			
DSCS III POWER SUMMARY	SUMMER SOLSTICE	WINTER SOLSTICE	DAYLIGHT	ECLIPSE BATTERIES	ECLIPSE BATTERIES	SURV. MODE
ATTITUDE CONTROL	62.2	62.2	62.2	62.2	62.2	62.2
PROPULSION	1.85	1.85	1.85	0	0	0
TT&C	128.4	128.4	128.4	149.8	127.3	44.9
COMMUNICATIONS	479.3	479.3	479.3	479.3	479.3	0
EP&SD	4.0	4.0	4.0	4.0	4.0	4.0
GIMBAL ANTENNA	3.2	3.2	3.2	3.2	3.2	4.0
SCT	64.5	64.5	58.3	207.8	58.3	0
THERMAL	59.3	48.7	113.4	64.2	64.2	See Figure
SUBTOTAL	802.8	792.1	850.7	970.5	798.5	7.2
DISTRIBUTION LOSS	16.1	15.8	17.0	19.4	16.0	-
PRU STANDBY	10	10	10	0	0	-
TOTAL LOAD AT PRU	828.9	817.9	877.7	990.0	814.5	

7-2 RECHARGING/BATTERY CHARGING REQUIREMENTS.

The EPDS contains three identical nickel-cadmium storage batteries operating in parallel. Each battery is an assembly of sixteen nickel-cadmium cells connected in series. The ampere hour capacity of each battery is 32 amp hours (at 10°C). The maximum total depth of discharge during the battery cycle is estimated (at end of life -EOL) to be about 65 amp hours for all three batteries.

The battery is capable of accepting a maximum charge current of 8.0 amperes with the battery voltage limited by the Battery Charge Regulator (BCR) which is part of the Power Regulation Unit (PRU). While being overcharged the battery voltage is controlled by Figure 1. The PRU regulates the satellite power bus to 28v ± .28v over all satellite load conditions. Each battery is designed for 900 discharge-charge cycles within a range of 10° ± 10°C. The life capacity is based on a maximum discharge-charge cycle of the type described below:

- Nominal average discharge of 16.1 amps per battery for 1.2 hours.
- A maximum discharge of 18.4 amps (SCT active) for 1.0 hours. This maximum discharge is constrained to no more than 20% of the total battery discharge time over mission life.
- Minimum charge of 2.8 amps per battery for 22.8 hours per day with voltage limiting by the BCR.

The batteries are capable of supporting heavier discharge periods but this will reduce the battery life cycle capability.

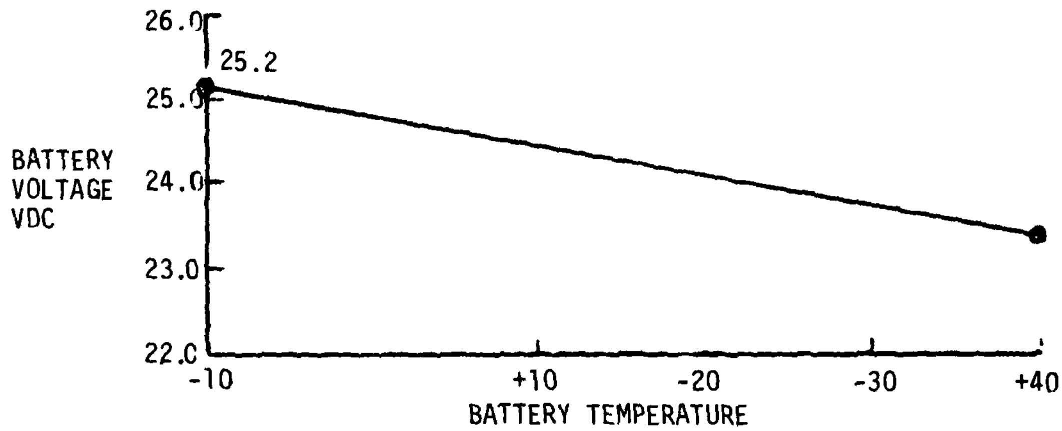

Figure 7-1. Battery maximum voltage.

The Battery Control Regulator(BCR)has three sources that limit Battery Charge Current. These are:

- High/Low charge rate resistive limiting
- Battery voltage-temperature limiting
- Absolute Battery temperature triggered cut off charging current

The PRU accepts battery voltages as shown in Table 2.

Table 7-2. Battery voltage at PRU interface.

Operating Mode	Current Per Battery (Amps)	Voltage at PRU Interface (Volts)
Charge	0 to 7.8	16 to 25
Normal Discharge	0 to 15	16 to 22

The battery is removed from the BCR stopping all charging if the Battery temperature exceeds 32°C. The Battery must then be commanded by discrete command to return to the BCR charge mode.

7-3 MINIMAL OPERATION MODE.

When on board anomaly detection circuits determine a loss of attitude control from the Earth Sensor or excessive battery drain (longer than 80 minutes) the EPDS configures the power distribution to Survival Mode. When the survival mode is triggered by the Battery Discharge Timer the following loads are automatically disconnected:

- Comm Subsystem
- SHF TT&C
- All non-critical control heaters
- SCT Subsystem

The S-Band Downlink is turned on and survival heaters located throughout the satellite are set to turn on typically at -18°C. The survival heaters prevent cold temperature damage or loss of operation of the equipment which would be needed to recover the satellite during survival mode operations.

There are two levels of heater control, the first level of thermal control is provided by heater circuits called control heaters. The control heaters maintain temperatures above the minimum operating temperatures for their associated components. These circuits are controlled by discrete commands regulated by thermostat signals. The second level of TC is called the survival heaters for which power is hardwired and controlled by minimum temperature level thermostats. All heater circuits are protected by over-temperature thermostats to prevent overheating. Figure 2 characterizes the profile of power load requirements for the thermal control subsystem in a survival mode in a geosynchronous space environment.

An 80 minute timer in the PRU times the duration of the battery discharge which is due to either a loss of sunlight on the SA or more system power consumption than the arrays can provide. Since the only planned long discharge time is eclipse, a discharge longer than 72 minutes is an anomalous operation. To minimize the risk of satellite loss due to loss of bus power, the survival mode is established to minimize system power consumption.

7-4 SPECIAL SXTF ISSUES.

There are several issues which, in the concept development of a non-interferring satellite powering/recharging system, are peculiar for use in the SXTF.

a) The satellite can operate for only a limited amount of time on battery operation. For DSCS III approximately 80 minutes of battery operation results in a 70% depth of discharge.

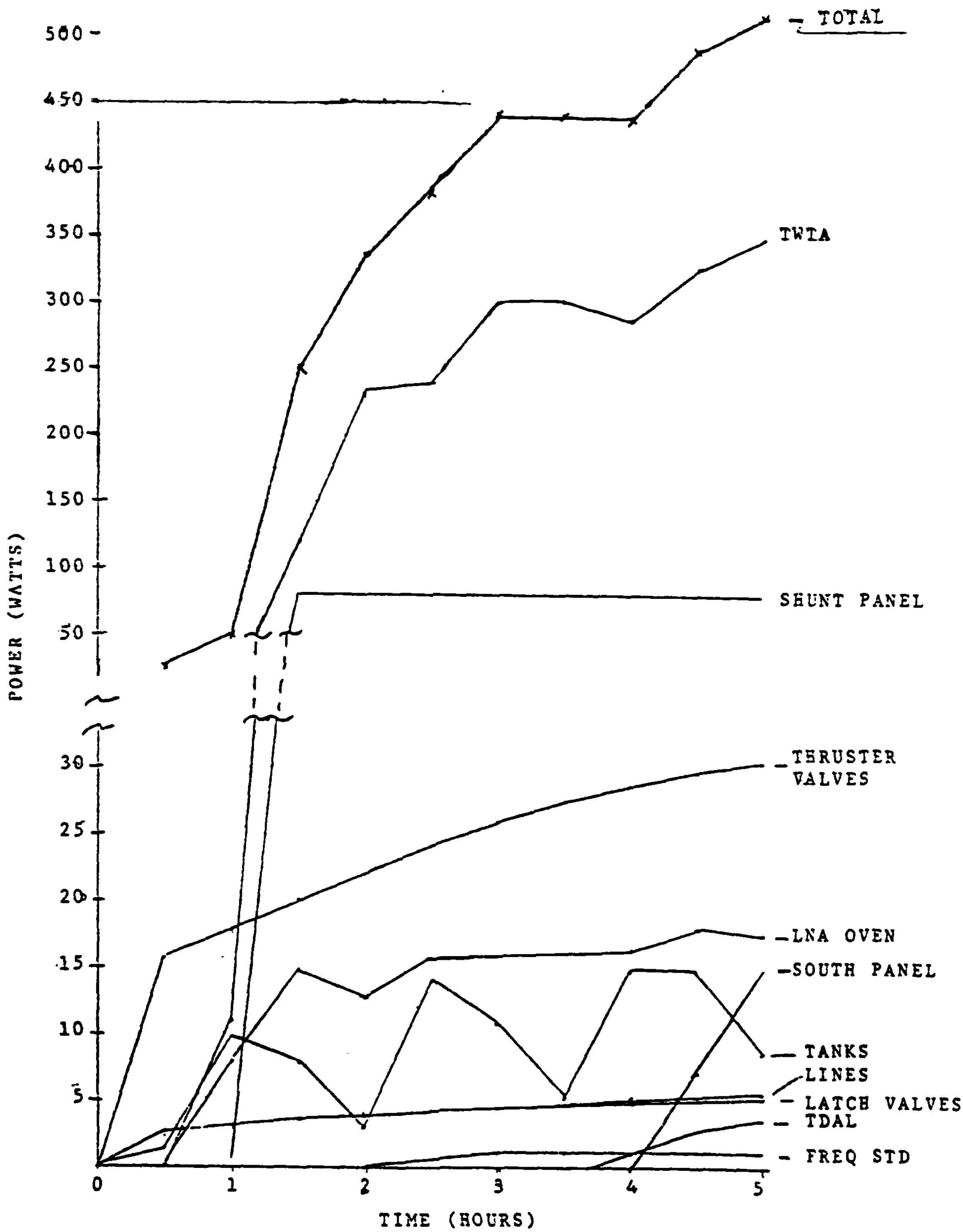

Figure 7-2. Heater power profile, survival mode.

b) In a thermal vacuum environment the presence of cryopanels at LN_2 temperatures prevents a partial subsystem operation scenario for very long to reduce the power consumption. The thermal control subsystem of DSCS III or any other satellite will automatically turn on survival heaters to maintain a minimum satellite temperature ($\sim$ -18°C). The maximum power consumption of these heaters is nearly equivalent to the power loads of the subsystems they are designed to protect.

c) The umbilical or isolated recharging/powering concept must <u>not</u> represent a possible single point failure in getting power to the satellite after the experiment. The time required to bring the satellite and cryogenic system back to room temperature from an LN_2 thermal environment may be on the order of 10 hours which is much longer than the probable life of the batteries with the satellite even in a survival mode. A detailed thermal analysis would be required to determine more definite times based upon actual satellite orientation and the proximity of the satellite to the MBS or PRS sources which represent warm spots in the chamber thermal environment.

d) The system must be such as to minimize any transients into the power subsystem during the make or break contacts because of any high voltage buildup between the satellite and ground due to satellite charging in an electron plasma environment.

e) The most appropriate position or connector for access to the satellite electrical power distribution system,EPDS,may not be very accessible or advantageous as a make/break point for an umbilical or powering/ recharging harness. Therefore some routing of a special harness or

connector cable may be required.

f) The orientation or routing of the harness between the EPDS connector and the make/break connector for electrical isolation during the test must be such as to minimize the structural or ground plane perturbations to the satellite and its SGEMP or spacecraft charging response.

g) The umbilical or power/recharging system must be made of materials and oriented so as not to interfere with or shadow the exposure of the satellite or have a significant interaction with the radiation environment, such as, charging of any insulating dielectrics in the electron plasma.

7-5 UMBILICAL SUBSYSTEM INTERFACES.

The Umbilical Subsystem interfaces with vehicle contractors equipment consisting of the satellite ground test connections on one hand, and the contractors ground test equipment on the other. The satellite side of the interface may in reality be EAGE, since protective components are frequently used at this point and in some cases may be under remote control. Figure 7-3 indicates the relationship. Internal interfaces of the umbilical are dependent on the facility structure and the suspension or related subsystems within the Chamber. Reliability is foremost in their selection criteria.

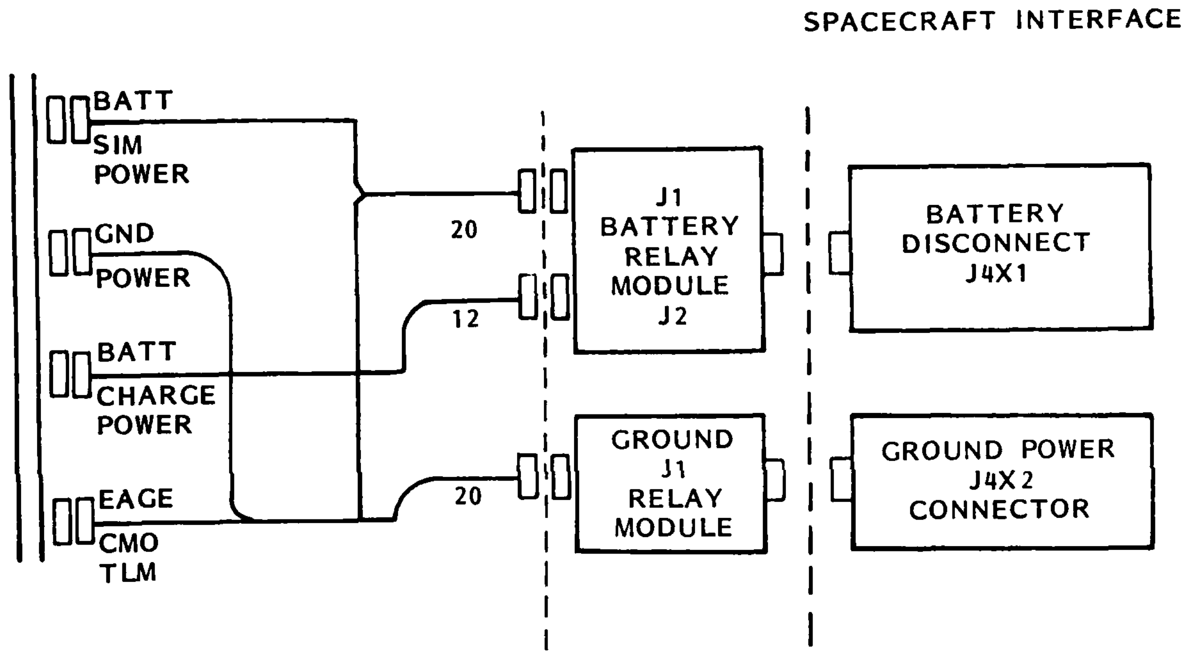

Figure 7-3. External cables vehicle—EAGE interface.

The EAGE attached directly to the satellite is a function of power subsystem size and complexity. A relatively simple system consisting of a few solar cells would require nothing more than a steering diode. The DSCS III system requires two relay modules, see Figure 7-3, to provide signal and power isolation from cable faults and to permit transferring satellite batteries to on-line or off-line states. The amount of EAGE in the equipment/control area is again a function of the satellite complexity, ranging from a single standard equipment rack, up to possibly as many as three or four racks. The adaptation of the EAGE to the umbilical is a contractor responsibility. It is desirable to not only select electrical interfaces for reliability and function, but also for flexibility and cost impact. Maintainability concerns further indicate use of an interface terminating system that is readily repairable. Maintenance time is not critical in view of anticipated scheduled maintenance flexibility in the overall SXTF sequence.

7-6 SATELLITE UMBILICAL REQUIREMENTS.

The following elements define the specific requirements on any umbilical system which reflects some of the issues already described.

a. Provide battery charging of the satellite from external source with appropriate monitoring, limiting circuits, and controls.

b. Provide powering the satellite from external power source for performance and/or functional testing.

c. Powering the satellite and charging the batteries may occur concurrently or separately.

d. Remote control of the disconnect/connect of all electrical circuits related to the satellite ground powering while satellite is at ambient or in vacuum environment.

e. Provide disconnection/connection to critical hardwired control and monitor functions appearing at the satellite to EAGE interface.

f. In the disconnected position, the umbilical subsystem should produce minimum interference with x-ray and related exposures of the satellite.

g. The subsystem should include the capability to safely discharge any residual charge remaining on the satellite structures, prior to reconnection of the umbilical.

h. The subsystem should function reliably in vacuum with minimal outgassing, and at ambient without special operating or service requirements. (Thermal control during vacuum operation may be required.)

i. Operational reliability of the subsystem is vital. In order to achieve a high degree of confidence in the subsystem operation, consideration must be given to the use of redundancy, cross-strapping techniques, alternate operating modes, and back-up equipment approaches. The remote control portion of the subsystem shall minimize the possibility of subsystem or damage failure dur to operator error, or failures in other facility subsystems.

7-7 UMBILICAL SUBSYSTEM CONCEPT.

Several approaches were evaluated for both the satellite-umbilical interface, i.e., supplying power to operate or recharge the power subsystem, and the actual satellite powering design. Table 7-3 describes several satellite powering concepts considered.

Conceptually, the Umbilical Subsystem is a transporter of power and signals between the test satellite and its EAGE, with the ability to effectively isolate the satellite electrically during exposure to the test x-ray spectrum. The Umbilical Subsystem as a facility item should be sufficiently versatile to accommodate a variety of test objects having external charging and/or powering requirements within the subsystem specifications. In considering approaches to achieve the desired capability, it is convenient to distinguish between two basic techniques:

a. Transfer of power and signals by conventional conduction and their interruption for x-ray testing by imposition of a dielectric between satellite and EAGE for isolation.

Or b. Transfer of power and signals by less conventional means, thus a dielectric of acceptable isolation properties.

In further evaluation of any approach, the relative magnitudes and frequency characteristics of the power and signals must be considered. For example, the maximum conceivable power transfer required for a DSCS III satellite would be 1.5 KW, essentially DC with a transient response capability of 1 millisecond for a 1/4 of full load step change. The signals transferred in either direction are DC, with a maximum source impedance of 10 kilohms. Response of transmission must be flat to 1 KHz. Signal amplitudes vary between 50mV and 30V. It is reasonable to assume that subsequent satellites will require power an order of magnitude greater than that required by DSCS III.

Table 7-3. S/C powering/isolation umbilical.

APPROACH	DESCRIPTION	ADVANTAGE	DISADVANTAGE
CONVENTIONAL UMBILICAL	MOTOR DRIVEN RAM WITH DIELECTRIC ROD FOR ALIGNMENT	PROVIDES A CONTINUITY BREAK-CLOSE TO S/C, HAVING MINIMAL EFFECT ON STRUCTURAL RESPONSE	DIFFICULTY WITH ALIGNMENT PARTICULARLY FOR S/C MOVEMENT BETWEEN SHOTS OR OVER LONG DISTANCES
PNEUMATIC DRIVEN-GENERATOR	AIR DRIVEN GENERATOR ATTACHED TO S/C CENTER BODY TO PROVIDE RECHARGING	GOOD ISOLATION INDEPENDENT OF ORIENTATION	ADDED WEIGHT AND VOLUME TO CENTER BODY MUST BE VACUUM COMPATIBLE MUST BE RADIATION HARDENED ASSOC. ELECTRONICS BE HARDENED
RF/MICROWAVE POWERING	TRANSMIT H.P.R.F. TO RECEIVER ON S/C AND CONVERT RF POWER TO CHARGE BATTERY	GOOD ISOLATION	HIGH POWER RF MAY BE SAFETY HAZARD - INTERFERE WITH SATELLITE COMM. SUBSYSTEM
STINGER WITH FERRITE ISOLATION	SIMILAR TO HURON KING EXPERIMENT	HIGH FREQUENCY ISOLATION	POOR LOW FREQUENCY/DC ISOLATION FOR S/C CHARGING
UMBILICAL - STINGER WITH FERRITE ISOLATION	COMBINATION OF APPROACHES ONE AND FOUR	ALLOWS SHORTER SEPARATION OF UMBILICAL BREAK AND REDUCES HIGH FREQUENCY COUPLING OF UMBILICAL LINE	MORE COMPLEXITY THAN EITHER APPROACH 1 OR 4 WEIGHT OF FERRITES MAY BE A CONCERN

Considering the nature and function of the signals to and from the satellite, it is apparent that although the quantity may fluctuate with the satellite program, the range of characteristics will probably remain unchanged.

There are a variety of techniques (other than conduction) for reliable exchange of signals in the range identified; however the movement of 1 to 10 kilowatts of DC Power in a controlled mode other than conduction is not quite so straightforward. As a consequence, it is delivery of battery charging and vehicle operating power that emerges as the predominent consideration with respect to an umbilical subsystem.

Elaboration on the ground power operating modes of a satellite will serve to indicate the major concerns in deciding between conduction and transmission of power to the test item. Table 7-4 identifies typical satellite power subsystem EAGE operating modes.

The sequence and frequency of satellite or test item powered operation within the SXTF impacts the approach and design of the umbilical subsystem. The operating modes in Table 7-4 can be related to a hypothetical test program or sequence to demonstrate the relationships and concerns. The general outline of such a satellite test program might include the following elements:

a) Installation and connection of contractor ground equipment in the SXTF.

b) Validation of all EAGE and facility equipment.

c) Installation of satellite in SXTF, and performance testing under ambient environment.

d) Pumpdown of chamber and satellite preparation for x-ray exposure.

e) Performance of x-ray exposures, with appropriate satellite functional testing.

Table 7-4. Spacecraft EAGE power characteristics.

SATELLITE OPERATING MODE	MAX VOLTAGE/ CURRENT TO SAT.	SPECIAL REQ'MTS AT SAT INTERFACE	POWER CONTROL CHARACTERISTICS AND FUNCTION
GROUND POWER (GP) (TURN ON)	20-30 VDC 0-4 ADC	NONE	- INITIALIZES SATELLITE POWER S/S, REQ'D FOR ENERGIZING SAT. FOR ALL GROUND OPERATIONS. - REQ'D FOR TRANSFERRING SATELLITE LOADS TO/FROM INTERNAL BATTERIES. - EAGE POWER SOURCE IS VOLTAGE REGULATED, TO WITHIN ±1%, WITH CONTROLLED RATE OF RISE AT TURN-ON. - PROVISION FOR OPERATOR CONTROL OF VOLTAGE WITHIN PRESET LIMITS (SOURCE IS PROTECTED AGAINST OVER VOLTAGE AND IS CURRENT LIMITED).
SATELLITE BATTERIES (OFF LINE/ON LINE)	N/A	REMOTELY CONTROLLED PWR SWITCHING CONTACT BUFFER RESISTORS REQ'D ON VOLTAGE MONITOR LINES.	- IN CONJUNCTION WITH GROUND PWR (ABOVE) ENABLES SWITCHING SATELLITE LOAD TO/FROM ITS BATTERIES. - BATTERIES ON LINE AND ADEQUATELY CHARGED, ENABLE ALL-UP OPERATION OF SATELLITE.
SATELLITE BATTERY CHARGING (SBC)	19 - 23 VDC 0 - 6 ADC	INTERLOCKED WITH POSITION OF CONTACT TO PERMIT CHARGING WHEN BATTERIES ARE OFF-LINE.	- ONE TO THREE BATTERIES ARE CHARGED, INDEPENDENTL - CHARGE CURRENT TAPERS TO ZERO AS BATTERY TERMINAL VOLTAGE APPROACHES PRESET LIMIT. - VOLTAGE LIMITING IS FUNCTION OF BATTERY TEMPERATURE (VOLTAGE LIMIT IS 15 ±0.1% OF VALUE). - CHARGING TERMINATES FOR BATTERY OVER TEMPERATURE - CURRENT LIMITED, OVER VOLTAGE/PROTECTED SOURCE.
SIMULATED BATTERY POWER (SBP)	19 - 26 VDC 0 - 50 ADC	TOTAL PWR SPLITS BETWEEN 3 NON-BUSSED CIRCUITS	- ENABLES ALL-UP OPERATION OF SATELLITE. BATTERIE MAY BE CHARGED CONCURRENTLY. SOURCE IS VOLTAGE REGULATED, REMOTE SENSING, CURRENT LIMITED, OVERVOLTAGE PROTECTED. VOLTAGE REGULATION IS TO WITHIN ± 1%.

f) Return of satellite & chamber to atmospheric ambient.

g) Performance testing of satellite to the original ambient baseline and/or special functional tests.

The umbilical system initial operation is in conjunction with the validation of the contractors equipment and facility related hardware/software. Assuming complexity equivalent to DSCS III, the umbilical would be subjected to interface testing and functional operation with dummy loads replacing the satellite in the chamber. In the case of the initial test of a particular vehicle, it is quite probable that the validation would extend to a trial pumpdown to demonstrate facility/EAGE capability to supply vehicle power, maintain thermal control, and to verify vehicle positioning capacity. During such operations the umbilical subsystem would be required to demonstrate redundant operating capability, emergency operating modes, and discharge capacity.

During satellite performance testing at ambient, the umbilical would enable systematic simulation of the operating sequences to be performed later in vacuum. A typical sequence with emphasis on umbilical manipulation might be:

a) Connect umbilical and charge batteries, while starting performance testing with simulated battery power (SBP).

b) Transfer to satellite batteries when fully charged, using ground power source for transfer operations.

c) Remove (disconnect) umbilical to complete satellite performance test and for isolation verification.

d) Reconnect umbilical, charge batteries, configure satellite for pumpdown using either ground power source or SBP.

e) Trickle charge batteries and maintain minimum satellite operating configuration during chamber pumpdown.

f) Perform abbreviated satellite functional testing with satellite on batteries and umbilical disconnected.

g) Reconnect umbilical and top off batteries with satellite on SBP until start of x-ray exposure test series.

h) Switch satellite to batteries using ground power source and start x-ray test sequence.

i) At appropriate points, dependent on battery capacity and satellite loading, testing is halted, the umbilical mated, and batteries recharged. Testing is resumed on completion of charge operations.

j) In the event of facility power failure, the umbilical must be immediately reconnected and electrical recovery sequences started to return the satellite to minimum operating configuration and to assist in maintaining thermal control.

k) Discontinuing the test operations for any appreciable duration while at vacuum will require umbilical connection and satellite operation from simulated battery power.

l) Return of vehicle and chamber to ambient would require umbilical connection, and satellite operation on SBP.

m) The conclusion of the test program is ambient performance testing with umbilical closed and operation from vehicle batteries and/or simulated battery power.

The preceding demonstrates the frequency of operation of the umbilical subsystem and infers its high reliability characteristic at both ambient and vacuum. Its operation is essential to the performance and safety of the test article while in the SXTF; this concept must dominate all design considerations.

Considering the power transmission through a dielectric approach, review of the control characteristics in Table 7-4 leads to the concept of bulk power delivery to special purpose conversion/control devices adjacent to the test item located in the chamber. That is, a single power generator, an RF source or pneumatic generator, for example, would transmit energy to a receiver on which multiple power regulators would be attached to perform the functions identified in the table. These regulators might in effect be the existing EAGE regulators currently used at ambient, or a unique design for operation in vacuum or under radiation exposure. With this concept the following factors become important:

A) Size and weight - The DSCS III equipment, for example, to provide satellite power and battery charging, weighs approximately 160 lbs. and occupies 16 cubic feet of space. A 10 KW system would be 2 to 3 times as large. A unique design regulator set would significantly reduce volume and weight.

B) Environmental Control - Using regulators designed for operation at atmospheric pressure requires their enclosure in a presurrized container with extensive thermal control to accommodate the convection cooling characteristics of the pressurized device. A regulator designed for vacuum and thermal radiation environment would overcome leakage and contamination concerns and would utilize the chamber wall thermal control capability.

C) Reliability - Ambient EAGE designs for satellite power are generally modular, permitting expeditious replacement of a major element in the event of internal failure. Redundant units are necessary to avoid test delays, and to insure safety in satellite power subsystem operation. Here again, the use of a unique design regulator would overcome many of the problems relating to using equipment designed for ambient use.

The use of the satellite's power regulation EAGE in the chamber in conjunction with a dielectric power delivery system has the advantage of minimizing facility changes between test vehicles. However, this approach imposes some strong limitations on the design of the power regulation unit mechanical configuration and remote control capability, for thermal vacuum operation. These limitations may infact require significant redesign of a satellite users EAGE. Summarizing, then:

1) The use of the satellite's EAGE power regulators in the chamber may preclude the need for an extensive generator design and development program to meet the variety of power subsystem requirements of different satellites, but it is undesirable from the reliability and efficiency stand point and may infact require new design efforts for each satellite system.
2) A specially designed regulator set unique for each test satellite or object would overcome objections to the use of the ambient EAGE in a thermal vacuum environment, but implies a long term commitment to power regulator design/development programs in support each satellite contractor SXTF user.
3) Power conduction through a retractable umbilical to the test object permits the use the contractors EAGE with minimal redesign required for operation in the ambient environment for which it was designed. In addition, no major failure modes associated with outgassing and contamination from a pressurized container are introduced into the system.

SECTION 8

SPACECRAFT ISOLATION - RF COUPLERS

8-1 INTRODUCTION.

Antenna couplers provide a means for commanding the satellite operation, monitoring the satellite housekeeping functions and exercising the communications payload without direct electrical hardwire connections to the satellite. In addition, the RF power is radiated from the satellite and remotely absorbed, rather than being dissipated in dummy loads at the satellite, thus helping maintain thermal balance of the satellite.

Antenna couplers or "hats" which surround the antenna and fully absorb all radiated power provide good RF isolation and are often used for full-up systems tests. In a thermal vacuum environment, however the hat type of antenna coupler is not suitable because of adverse effects on the thermal balance of the satellite. Thus a more remote antenna coupler design is required for this type of application. One type of remote antenna coupler design is the oblique absorbing panel, which is being used presently in thermal vacuum tests of DSCS III. Similar RF probes are designed into both types of antenna coupler to form RF links with the satellite. These radiative RF links have performed successfully in system and thermal vacuum tests on the DSCS III communications satellite.

Presently military communications satellites such as DSCS II and DSCS III employ primarily UHF, S-band, and X-band frequencies. Other satellites such as those used for maritime and commercial communications employ L-band, C-band and Ku-band frequencies. Future satellites presently under consideration are likely to operate in even higher frequency bands 20, 30 and 60GHz.

The absorbing part of an antenna coupler or hat is generally designed to cover several bands, but some types of design cover only one band. The RF probes designed into an antenna coupler may likewise be either multi-band or single band depending on their type and the test requirements. A set of general purpose antenna couplers and probes that cover several frequency bands are suggested for SXTF use, with special antenna couplers and probes tailor made to meet specific satellite test requirements.

Not only do the TT&C and communications payload antennas on a typical satellite range from UHF to X-Band and above, but they also vary in their directional characteristics from very narrow beam to broad beam earth coverage to hemispheric coverage. Another factor affecting the RF coupler design is the radiated power level from the satellite antenna, which has an effect on the power dissipation properties required for the coupler absorber material, the thermal transfer characteristics required for the coupler and the power levels reflected into the chamber.

8-2 ANTENNA COUPLER/HAT DESIGNS.

Three different antenna test hat and coupler designs have been evaluated for satellite system and thermal vacuum tests. These are referred to as RF Hat designs A, B, and C. The designs are distinguished primarily by their absorber properties, form factors and applicable frequency bands. Hats A and B both use broadband pyramidal absorber material fastened to a metal backing while Hat C uses a flat conductive-sheet absorber spaced above a ground plane by a layer of low-density foam material. RF probes designed for specific frequency bands are integrated into each antenna coupler/hat for coupling to the various satellite antennas.

A sketch of RF Hat design A is shown in Figure 8-1. This RF coupler is lined with 18"-long absorber cones and covers UHF, S-band and X-band with a relatively small mismatch (VSWR about 1.2 to 1.3). The RF probes are imbedded in the absorber material. They consist of a dipole over a ground plane for UHF, crossed dipoles over a cupped ground plane for S-band, and open-ended WR-112 waveguide for X-band. Several X-band probes are located so as to be on-axis with one of the several DSCS III X-band antennas when the satellite is positioned under the RF hat during systems tests.

RF Hat design B is shown in Figure 8-2. This antenna coupler has the shape of a rectangular box with one face open and provides more effective shielding and absorption of the RF radiation from the satellite. Because of its enveloping shape this design is not as appropriate for thermal vacuum applications. Hat B is lined with 8"-long absorber cones, which cover S-band and X-band very well but not UHF. The RF probes for Hat B are implemented only for X-band and consist of eight high-performance circularly-polarized horns. These horns are located so that each is on-axis with one of the eight DSCS III X-band antennas when the hat is aligned with the spacecraft. During system tests with this RF Hat the RF links are used only at X-band, and the S-band and UHF signals must be hardwired with coaxial cable.

RF Hat design C is shown in Figure 8-3. This consists of a large ground plane of lightweight honeycomb construction over which a flat conductive sheet of absorber is properly spaced at X-band by a layer of low-density foam material for an oblique incidence angle of 40°. Such an absorber panel has a much narrower bandwidth than the pyramidal absorber material used in Hats A and B, but it is more suitable for the thermal vacuum tests because of its lower outgassing characteristics. The limited bandwidth of Hat C

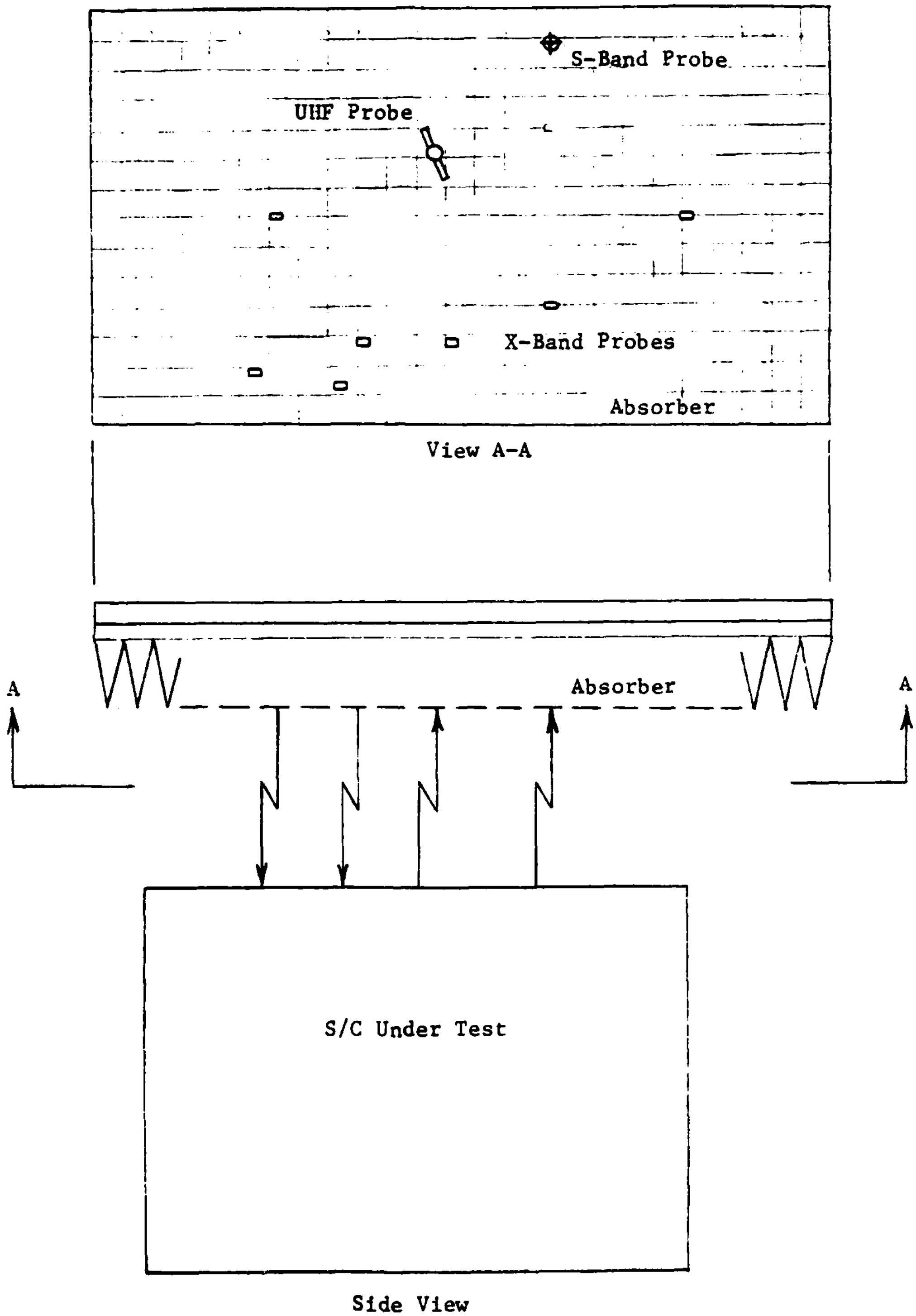

Figure 8-1. Antenna coupler/RF hat—Design A.

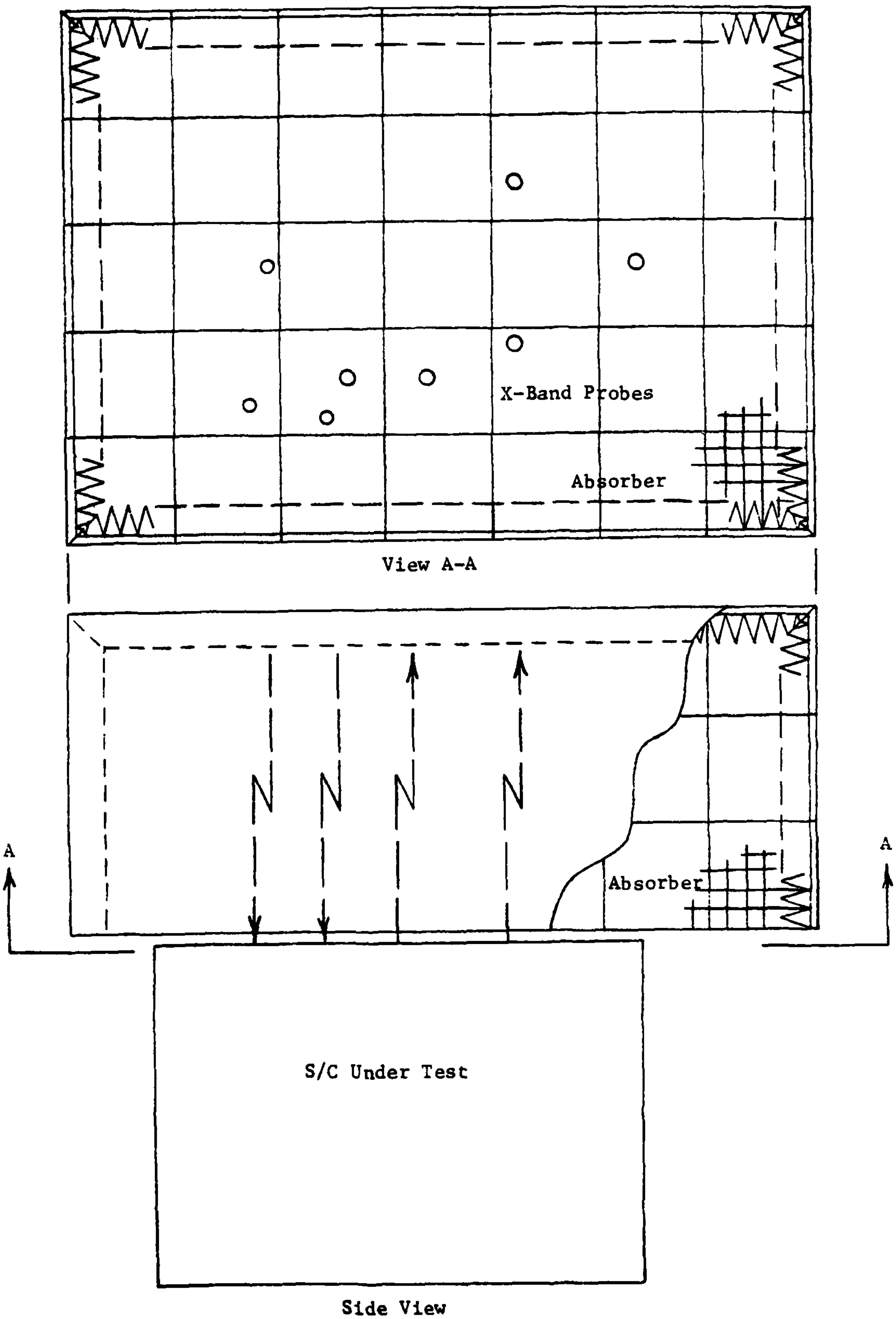

Figure 8-2. Antenna coupler/RF hat—Design B.

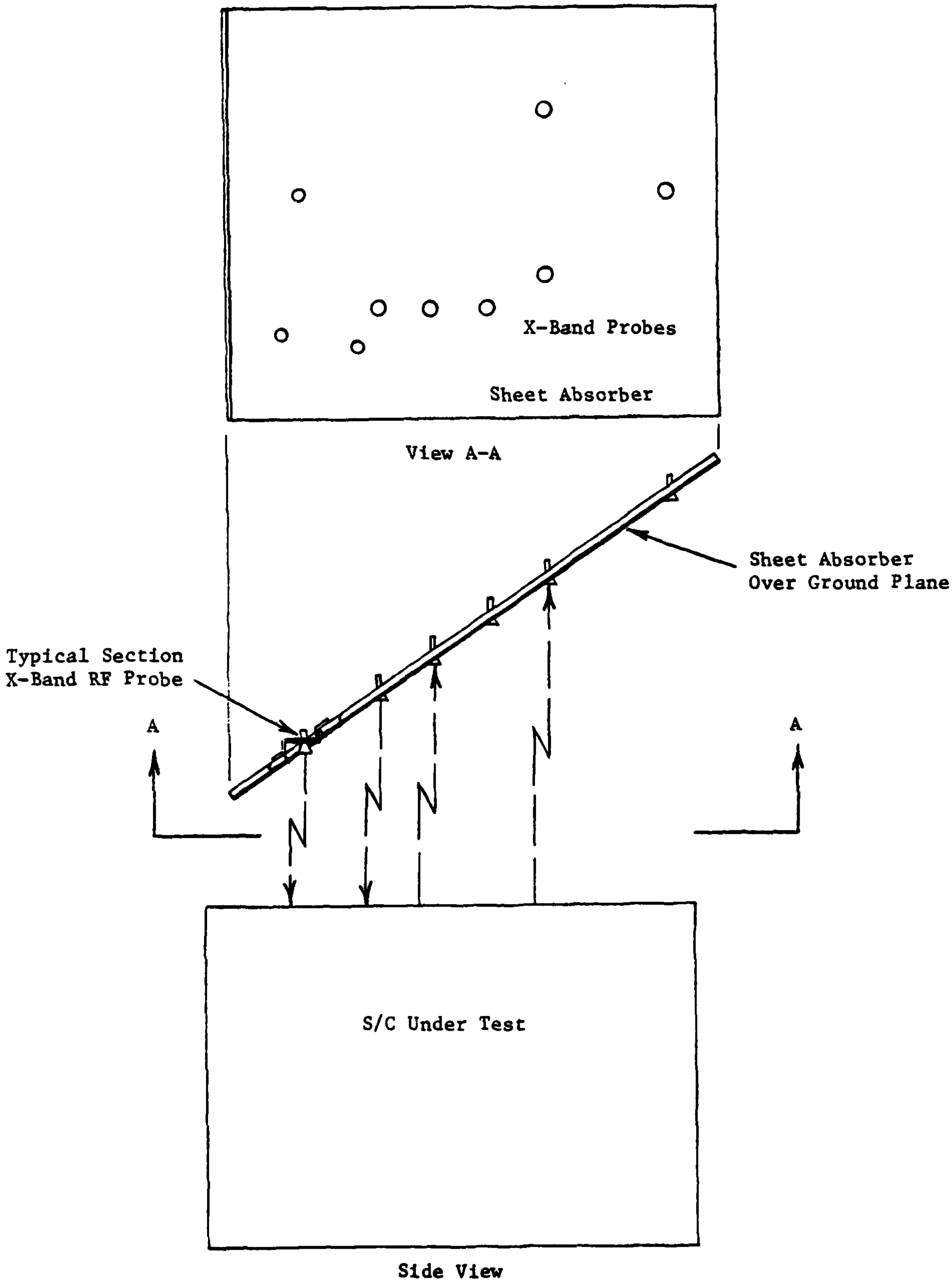

Figure 8-3. Antenna coupler/RF hat—Design C.

is acceptable for the SXTF test if only X-band use is required. As with Hat B, the RF probes are implemented only for X-band and consist of high performance horns. The horns are mounted on brackets at the 40° incidence angle so as to be on-axis with the spacecraft antennas. By inclining the plane of the coupler at an angle of 40° to the satellite antenna axis, any power rejected by the coupler due to residual mismatch will be directed away from the satellite. This reflected RF energy is then dissipated by internal reflections within the vacuum tank as discussed in Section 8.4.

The basic design principles of the A, B, and C RF Hats have been successfully demonstrated during tests with the DSCS III Communications Satellite and are well suited for the command and monitoring applications. For application to the SXTF, however, these designs must be extended to provide more versatile general purpose antenna couplers with wider bandwidths and adjustable RF probes. Each antenna coupler should cover several frequency bands, and the absorber materials used should be both broadband and suitable for the thermal vacuum environment. Sets of interchangeable RF probes would be desireable to permit the antenna coupler to be adapted to different combinations of frequency bands and antenna locations that are unique to particular satellites that may be tested in the SXTF.

8-3 WIDE BAND DESIGN CONSIDERATIONS.

Wide-band antenna couplers which cover several communications satellite frequency bands are desireable for the SXTF so as to provide a versatile general purpose facility. The absorber materials for such antenna couplers should be well matched over a wide frequency band and must be compatible with the thermal and vacuum environment. The RF probes for each particular frequency band must be designed for low reflection, proper polarization, and appropriate beamwidth and gain. Ideally, the RF probes should also be

adjustable and interchangeable so that the overall antenna coupler can be reconfigured to match a variety of satellites to be tested. In practice, a trade-off between versatility and complexity is required which may result in a few sets of relatively simple antenna couplers.

With regard to the overall SXTF test configuration, a variation of RF Hat design A shown in Figure 8-1 appears to be the best basis for the design of SXTF antenna couplers. RF Hat design B shown in Figure 8-2 would be best for absorbed energy and minimal chamber RF interaction but it is not well suited to the thermal test environments and would shadow the test radiation exposure. RF Hat design C shown in Figure 8-3 provides better configuration but it is not sufficiently broadband for many applications.

An alternative antenna coupler design which combines features of designs A and C is shown in Figure 8-4 as design D. This variation of design A uses broadband absorber material with the panel tilted at a moderate angle to further minimize the residual reflection back to the satellite. The RF probes are mounted in the panel at the same tilt angle so as to be aligned with the satellite antennas as in design C.

The absorber material used in the antenna coupler must meet several rather stringent requirements for RF and thermal vacuum performance. It must have:

(1) Very low RF reflection back toward the satellite antennas over the frequency band of operation.

(2) Relatively low RF reflection and/or scatter levels in all other directions;

(3) Low outgassing in the thermal vacuum environment of 10^{-6} Torr pressure, -190°C, and high thermal radiation (high emissivity).

(4) Stable RF performance in the Thermal Vacuum environment.

(5) Relatively easy reconfiguration for various satellite antenna types and arrangements, and

(6) Small cross-sectional area to minimize shadowing the satellite from its thermal or test radiation environment.

The specific absorber material and shape will depend upon the frequency band to be covered as well as the above factors. For operation down to UHF, the absorber will be fairly thick, such as the 18"-long pyramidal absorber cones used in RF Hat design A, while for operation down to only S-band or L-band the 8" long pyramidal absorber cones used in RF Hat design B would be adequate. For operation only at X-band and above absorber cones only 3" or 4" long are sufficient. In any case, several different kinds of absorber material are available. It is critical that the absorber material be compatible with the vacuum and thermal environment.

Another approach would be to use a set of two or more different coupler designs, for example, a satellite oriented X-band coupler and a small broad beam S-band coupler designed to provide an RF link with the TT&C subsystem antenna. Some relaxation of the frequency-band requirements for the RF coupler could also be achieved by using fiber optics to support the UHF link between the satellite and the SXTF chamber. In that case, however, any transmitted UHF high power must be absorbed in a dummy load located at the satellite, which would change the thermal balance conditions from that for normal operation.

8-4 CHAMBER EFFECTS.

Not all of the RF power radiated from the satellite antennas will be absorbed by the antenna coupler, nor will all of the RF power radiated to the satellite antennas from the antenna coupler be absorbed. Some of this RF power will be reflected by either the antenna coupler or satellite at a low level, while some will not be intercepted and will thus spill over past

the antenna coupler or satellite. In either case, the SXTF chamber will tend to be excited by this RF power and perform like a hugh RF cavity.

Fortunately, the chamber is many wavelengths in length and diameter even at UHF and has many lossy obstacles present, such as the cryo-panels, EM damper curtain, support structure, etc. The radiated waves will tend to reflect between the walls, scatter, and decay rapidly.

In principle, however, higher order modes of excitation in the SXTF chamber could occur for some unique configuration of the test set-up and thus have adverse effects on the RF links. This could result, for instance, in transmission ripples over a communications frequency band or in one or more sharp dips in transmission in the critical TT&C band.

To assure that such adverse effects do not occur during the SXTF tests, it is important that exploratory swept frequency measurements and adjustments if necessary, be made for each planned test set-up prior to the actual system tests. This could be done with the antenna couplers in the planned configuration and a simulated satellite having antennas equivalent to the actual satellite antennas to be tested. Provision of RF test equipment for these measurements, such as swept oscillators, spectrum analyzers, etc. will thus be essential for the SXTF.

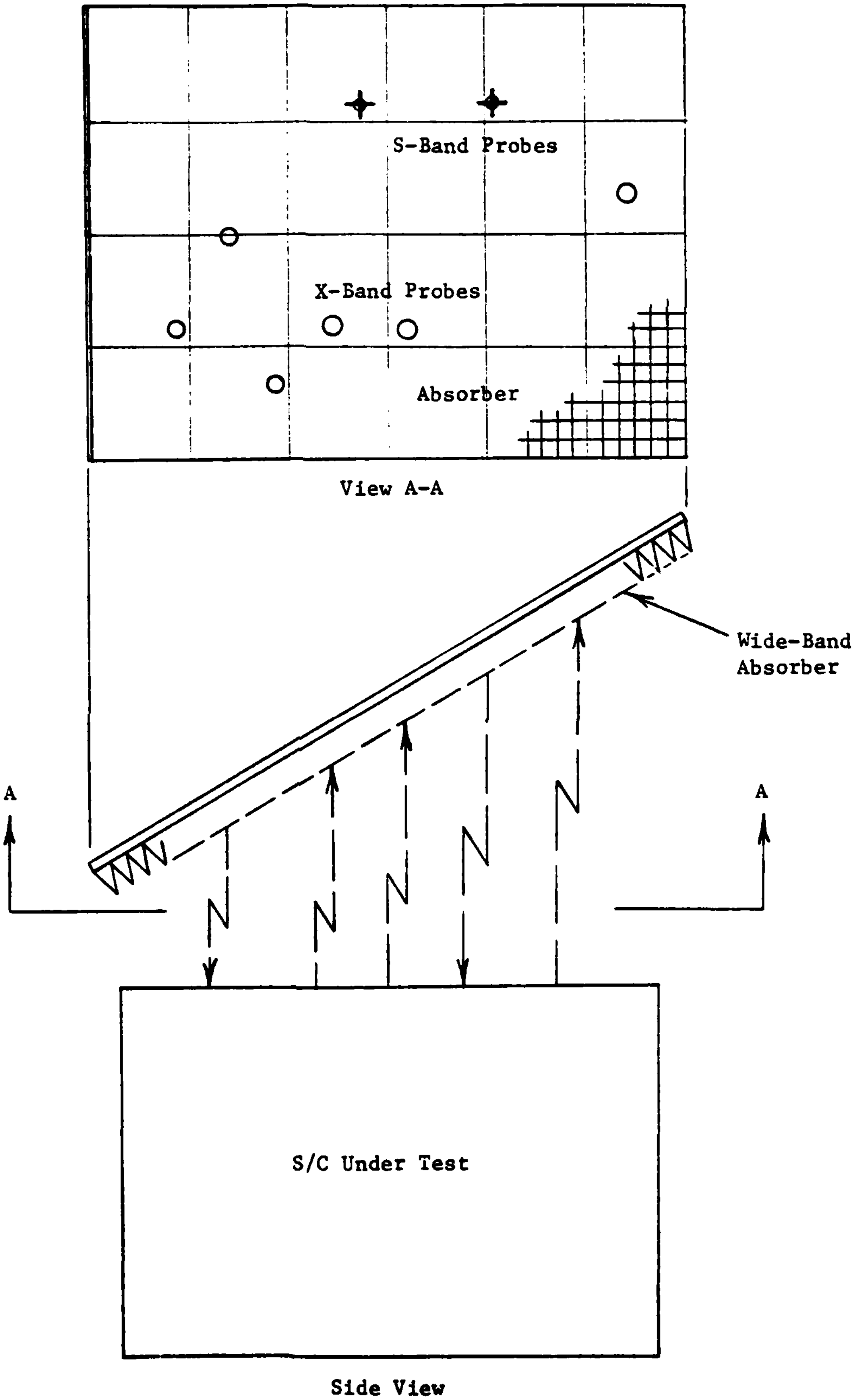

Figure 8-4. Antenna coupler/RF hat—Design D.

SECTION 9

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Spire Corp
ATTN: R. Little

TRW Defense & Space Sys Group
ATTN: D. Kingsland

TRW Defense & Space Sys Group
ATTN: P. Chivington

DATE
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